

89297 AN-3285  
PROCE201N

## SESSION A

# DAMAGE TOLERANCE IN HELICOPTERS - SITUATION TODAY

University of Cranfield, UK.

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## **SESSION A**

### **THE MANUFACTURERS PERSPECTIVE**



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**AGUSTA**

**FLAW TOLERANCE / DAMAGE TOLERANCE**

**AGUSTA EXPERIENCE**

**ON**

**METAL PARTS**

**Ugo Mariani**

**Chief Fatigue Office**

Past applications of Damage Tolerant approaches were related to support service bulletins.

Typically inspections were related to:

- easily detectable cracks, like visual detection of trailing edge cracks in rotor blades during routine inspection

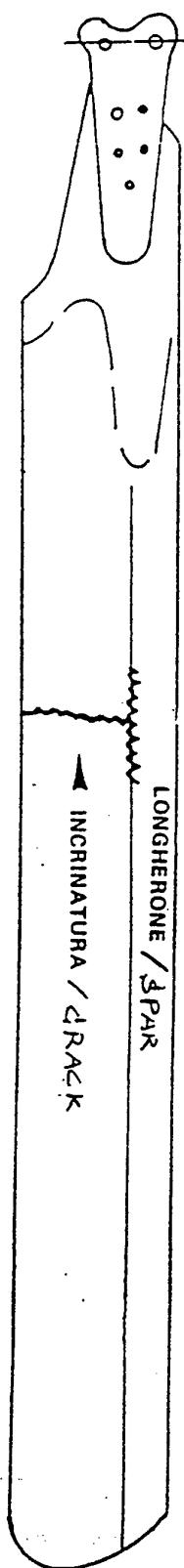
or

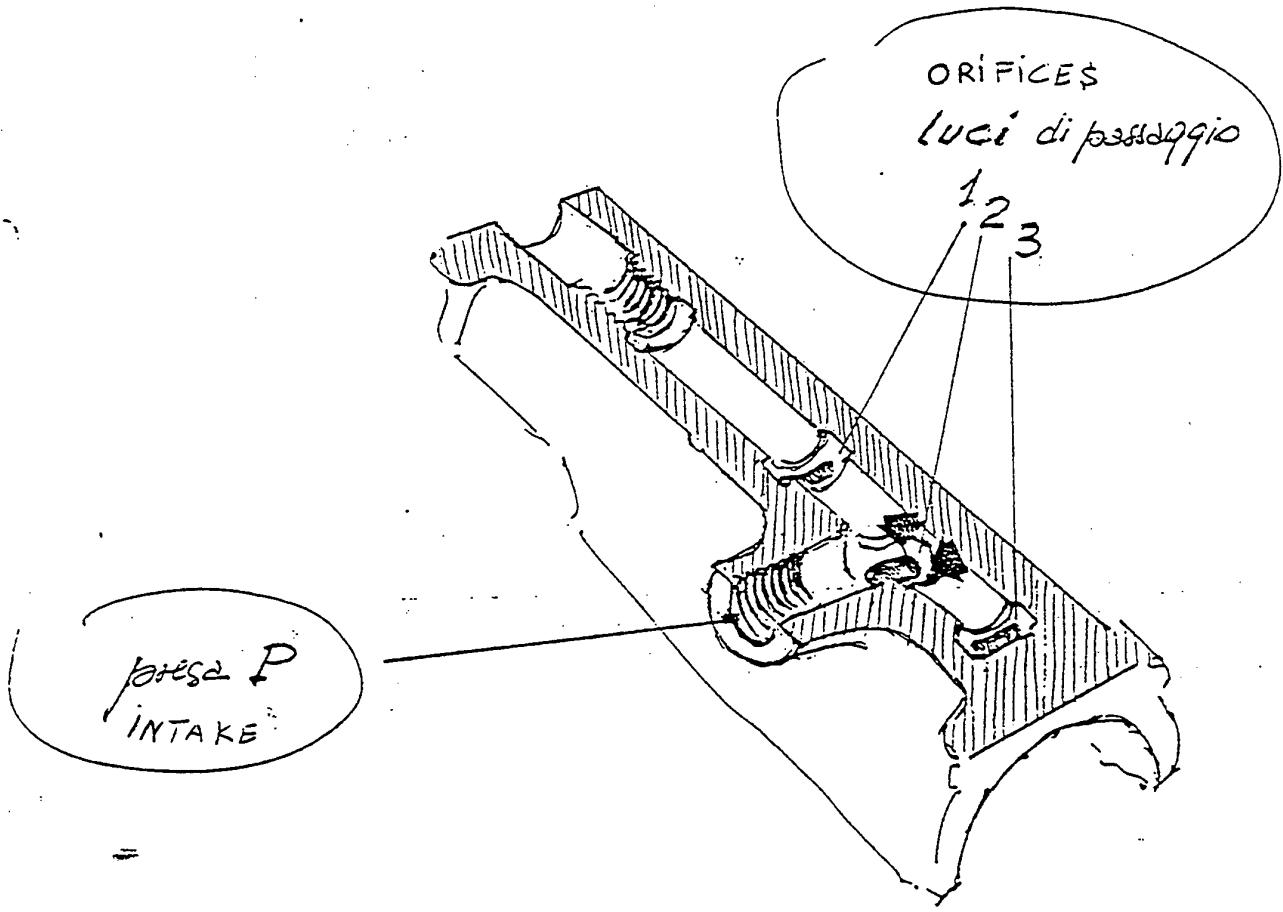
- self evident partial failures, like cracks in servoactuators, self evident due to hydraulic fluid leakage

More complex inspection methods were used for specific critical sections, focusing detection methods in a very limited area of the part.

In these cases frequent inspections could be accepted due to the 'short' time required.

This cannot be the general case for rotor or mechanical parts of a new helicopter.





See A - A

**FLAW TOLERANCE - ENHANCED SAFE LIFE**

Some experience was gained in EH101 development program and research activities.

The approach is now applied to NH90 and AB139.

A proper validation of the type of flaws and the method used to test flawed elements is considered essential.

An example is provided detailing the evidences to validate scratches in aluminum.

A survey of service data and test evidences provided information on a severe manufacturing scratch, showing that plastic deformation and notch radius at crack tip are the relevant parameters for the crack initiation.

The crack growing path of a component failed in service showed negligible effects of the notch were plastic deformation had occurred, although the crack was growing very close to the scratch.

Milling an 'U' shaped gauge in SCT specimens tested at CAL made a very sharp scratch. Considering the scratch like a crack,  $\Delta K$  was computed for each coupon. Depending on the applied stress, specimens failed if the  $\Delta K$  was higher than the threshold value, and they gave run out if lower.

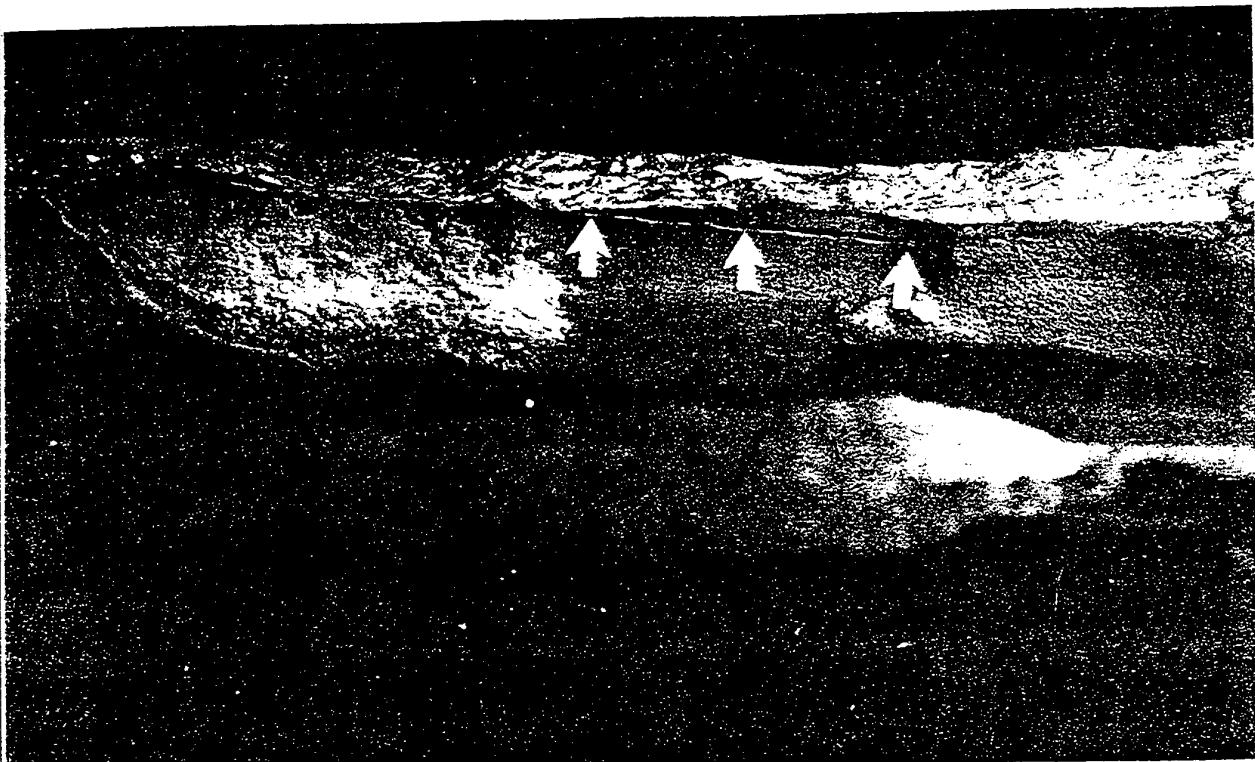
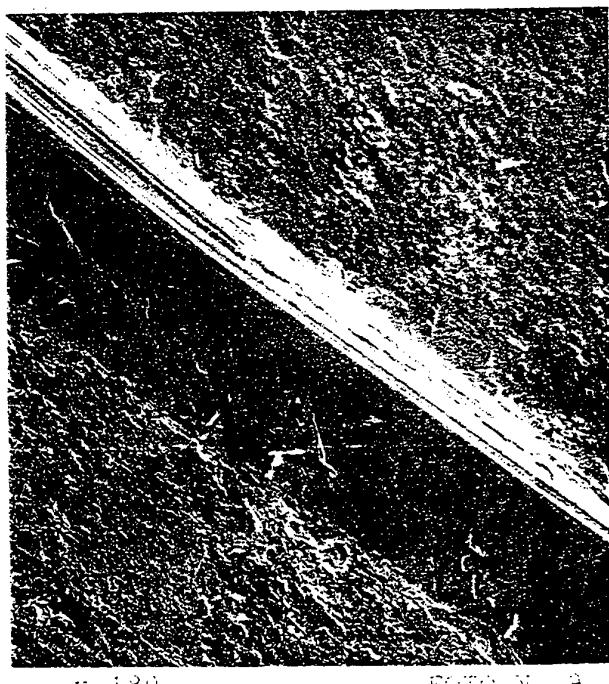


FOTO N. 7



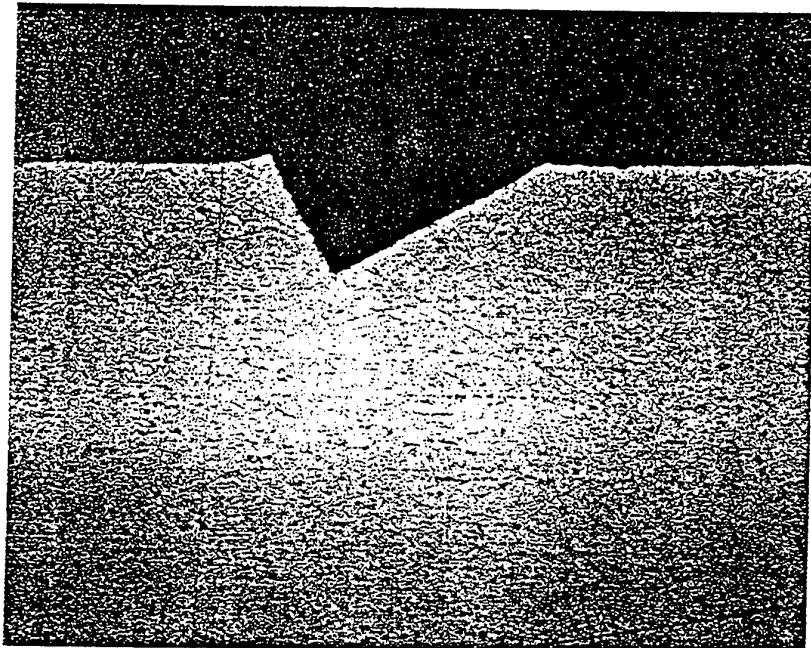
x 9

FOTO N. 8



x 180

FOTO N. 9



- LA PROFONDITÀ DELL'INTAGLIO È DI MM 0,15 ~.  
NON SI NOTANO INCRINATURE.  
L'ANGOLÒ DELL'INTAGLIO È DI ~ 90°

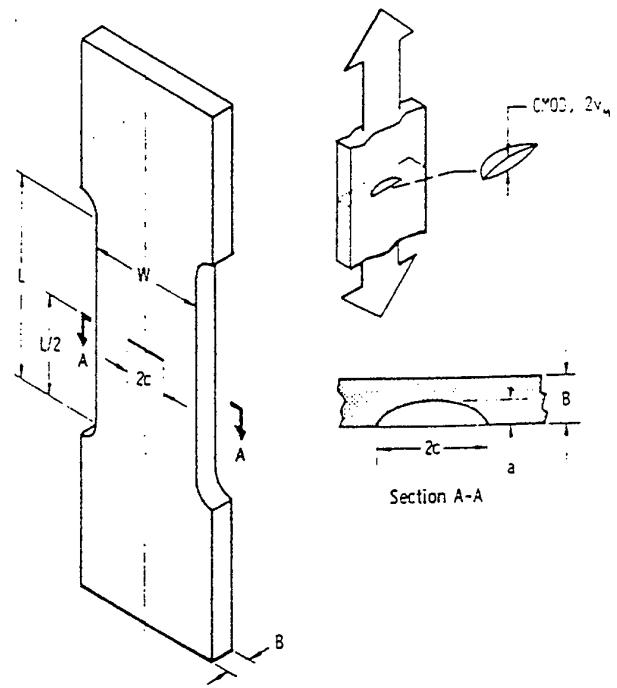
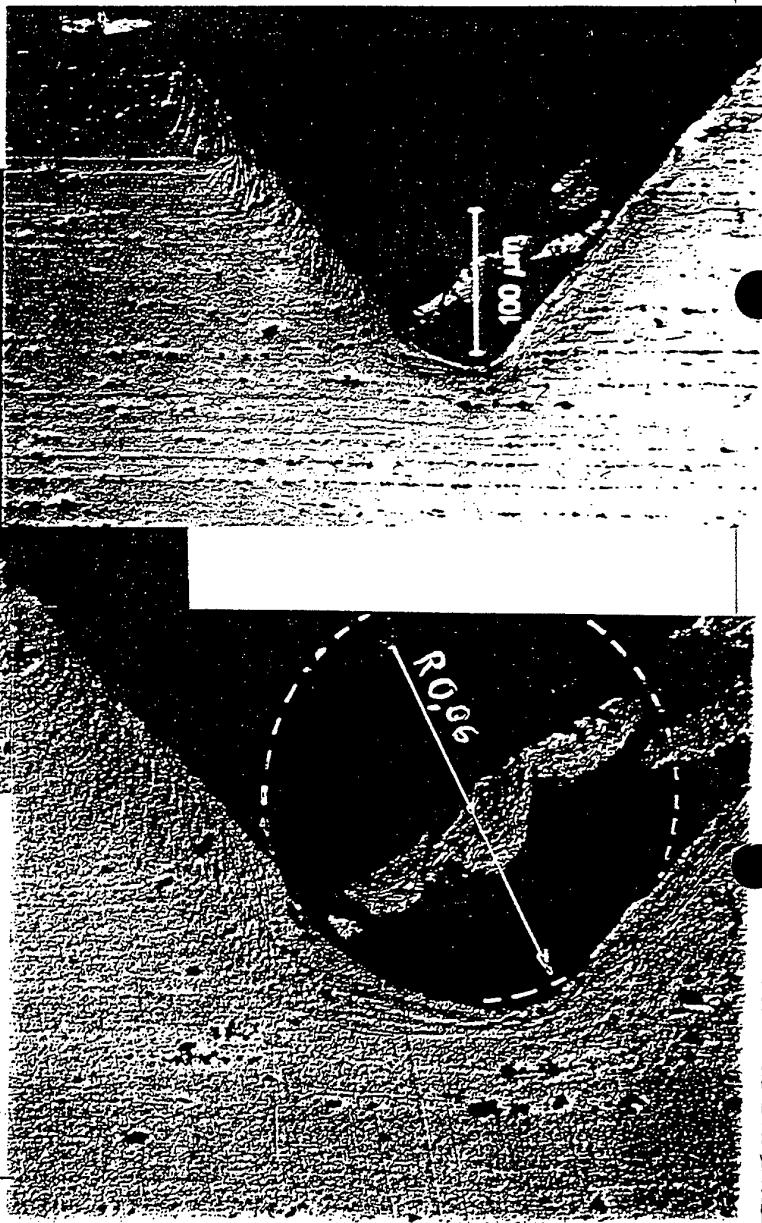


FIG. 1 Typical Surface-Crack Specimen (Grip Details Omitted) and Nomenclature

Foto n° 8 200X



Foto n° 9 500X



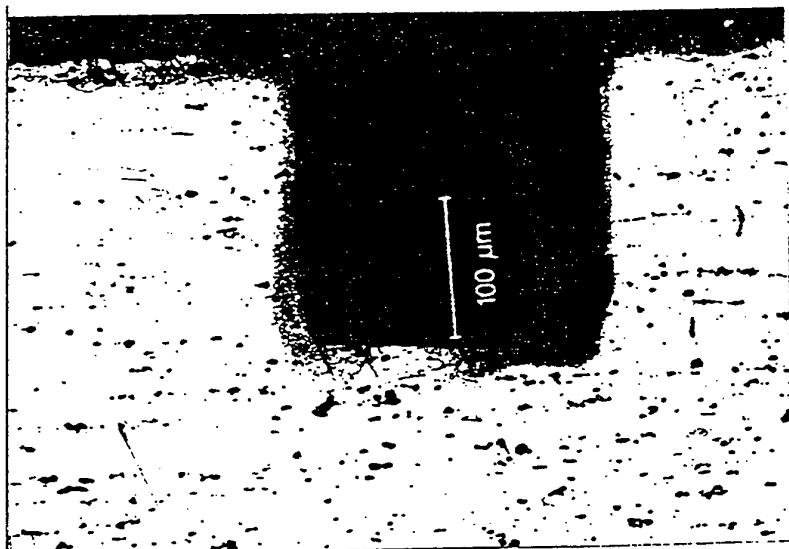
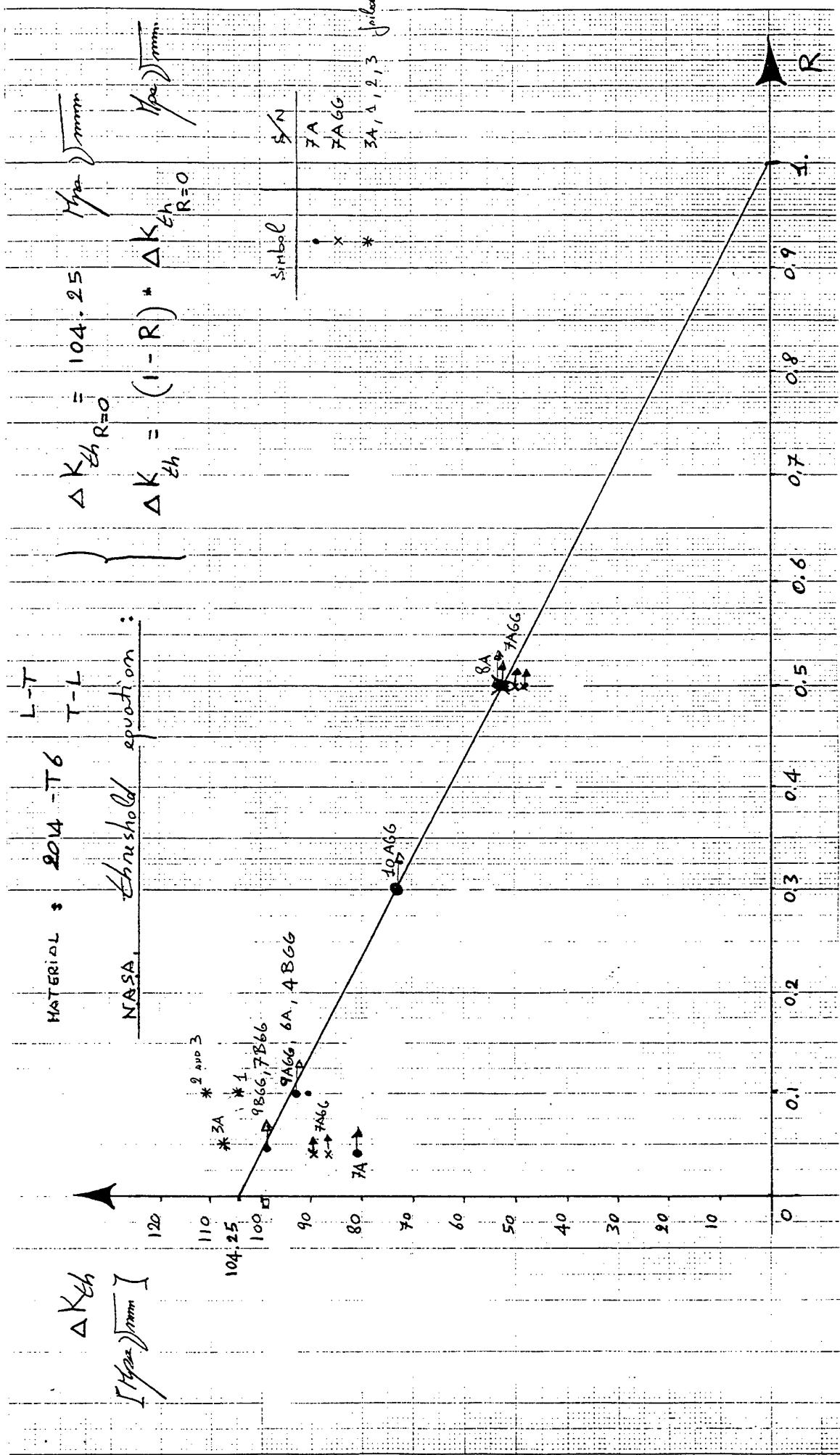


Foto n°1  
200X

Foto n°2 500X





Based on these evidences a proper tool could be selected to flaw a Main Gearbox Case, made by aluminum casting, to carry out a flaw tolerance validation.  
This is a 'worst case' of scratch, severe but still realistic, since it has:

- 'V' shape according to experience
- minimize the plastic deformation at the tip, shown by etched metallurgical section
- 0.5 mm depth, highest value consistent with experience

A block-loading test was carried out representative of 600 h, factored for scatter, considering 0.5 mm deep scratches as Clearly Detectable Flaws.

Failure occurred during the second test phase that was carried out to improve the result.

The flaw tolerant evaluation, based on 'time to crack initiation', will be used to support inspection intervals for accidental damage.

Assuming this as the retirement life of all the parts would be and unnecessary penalty since the flaws addressed are visually detectable during a Detailed Inspection.

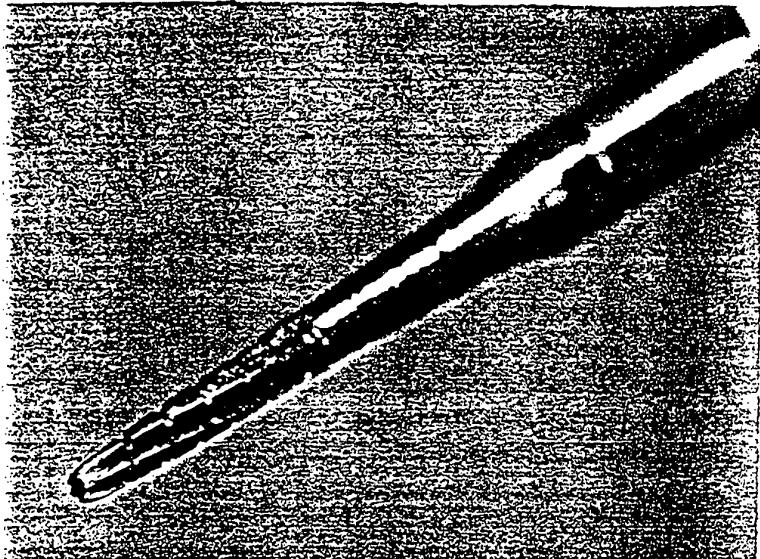


Foto n°18 10X

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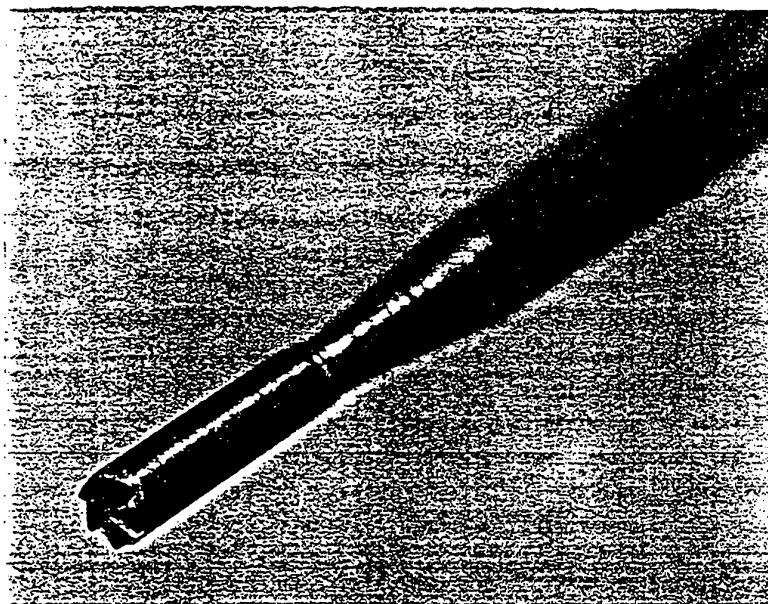


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sulla testa.

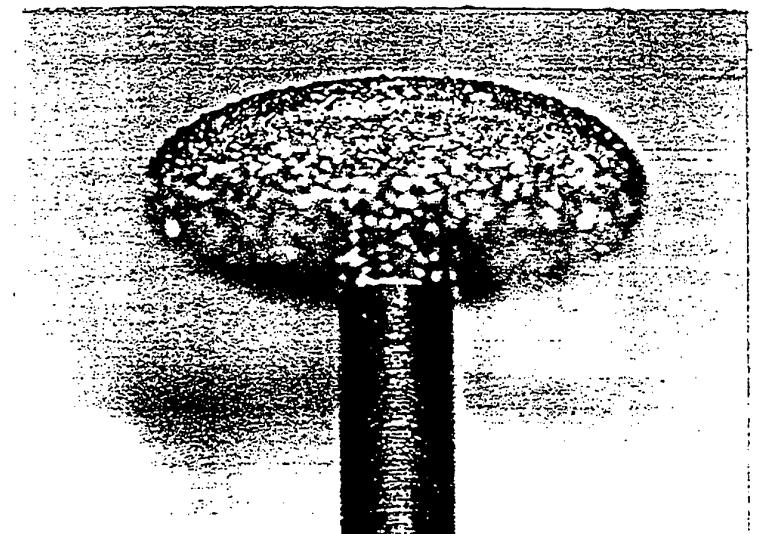
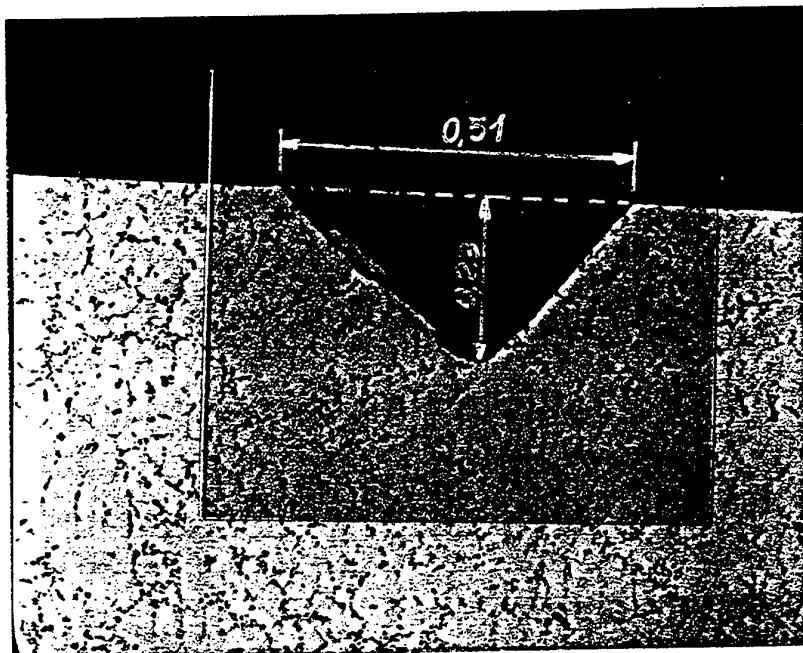
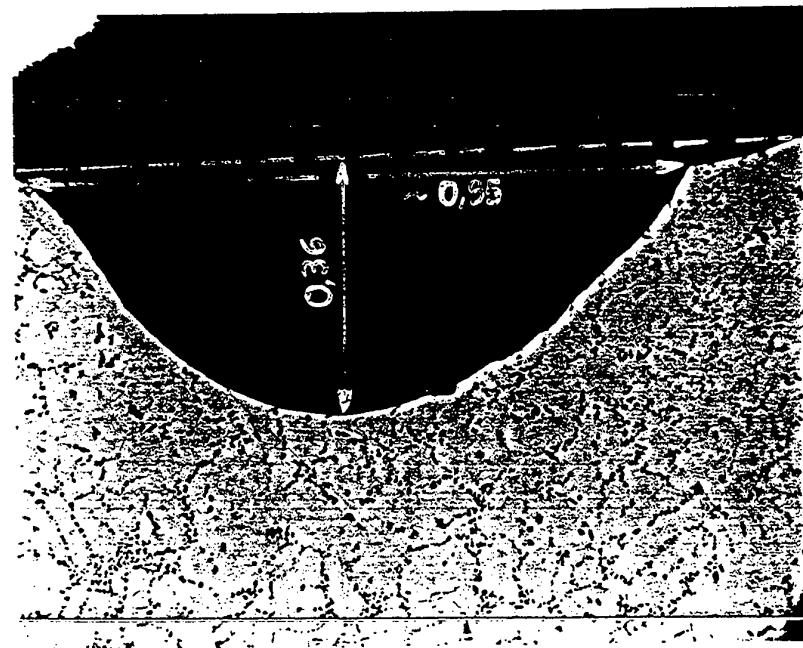


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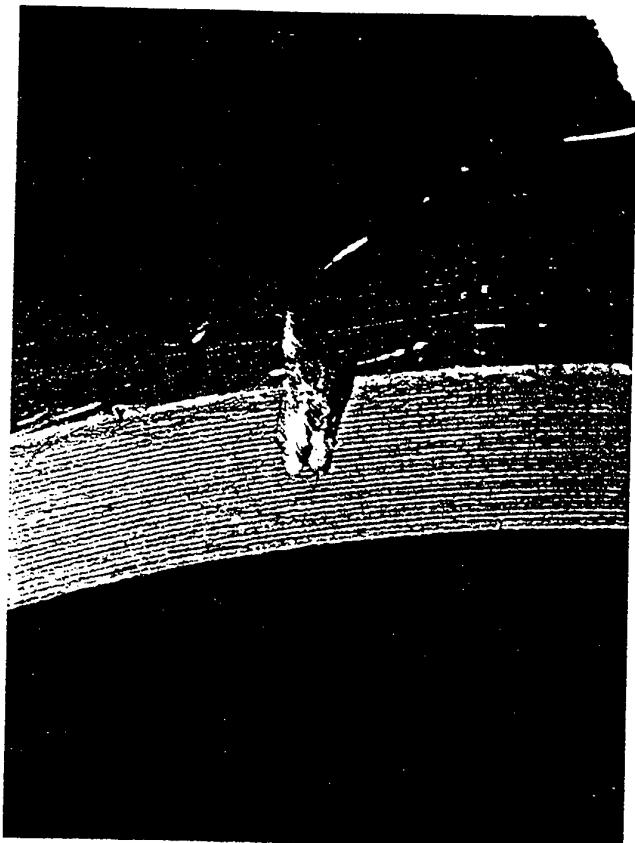
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diamantato,



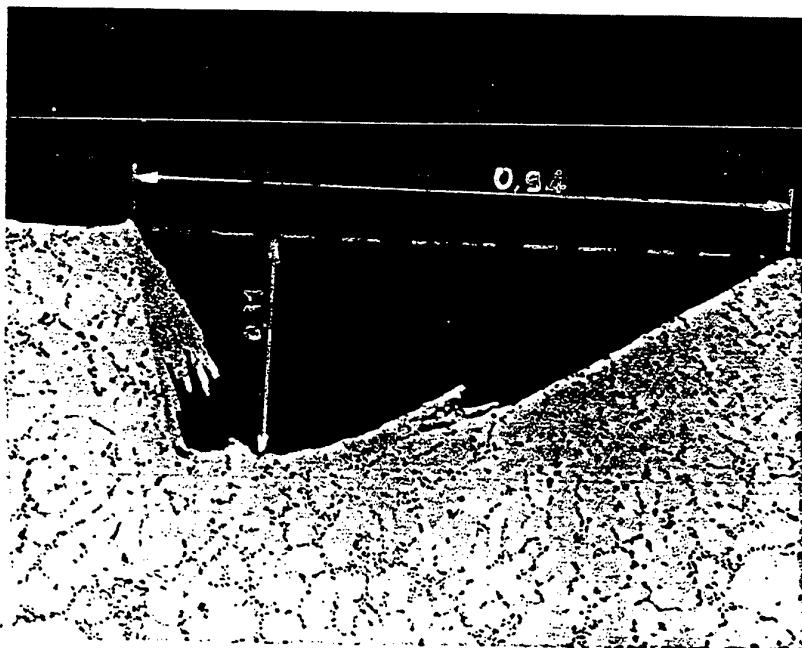
100X



100X



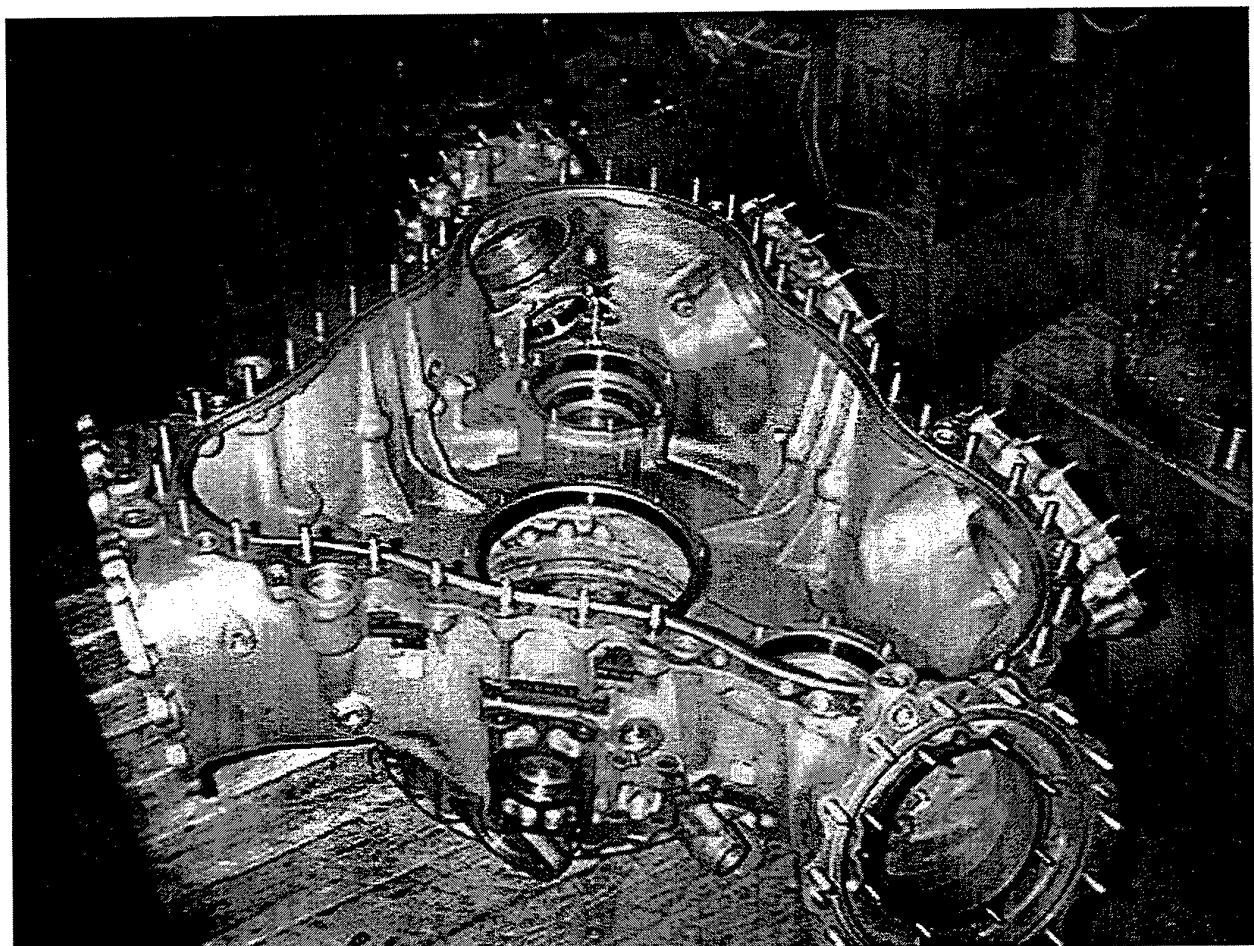
10 x



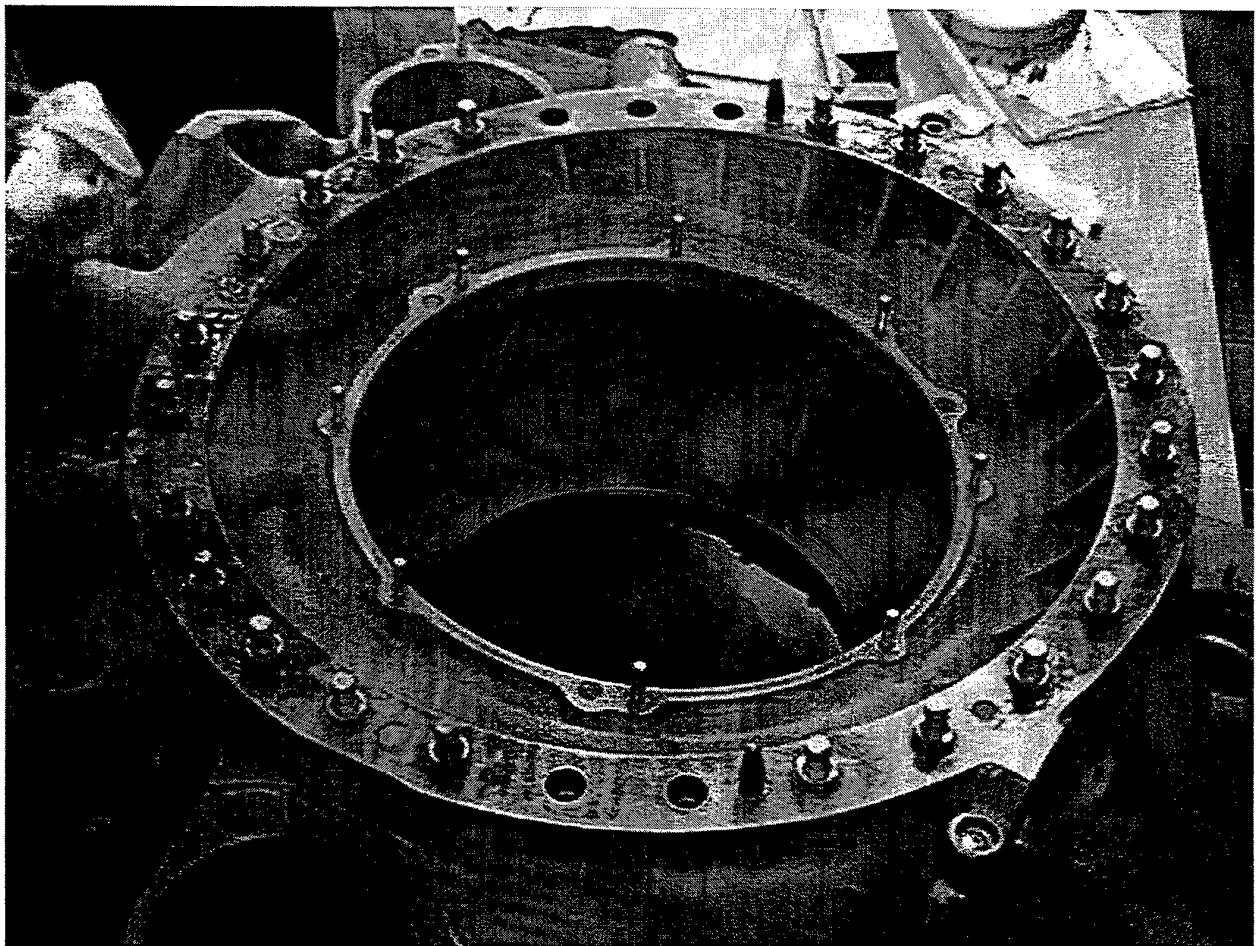
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S/N 3

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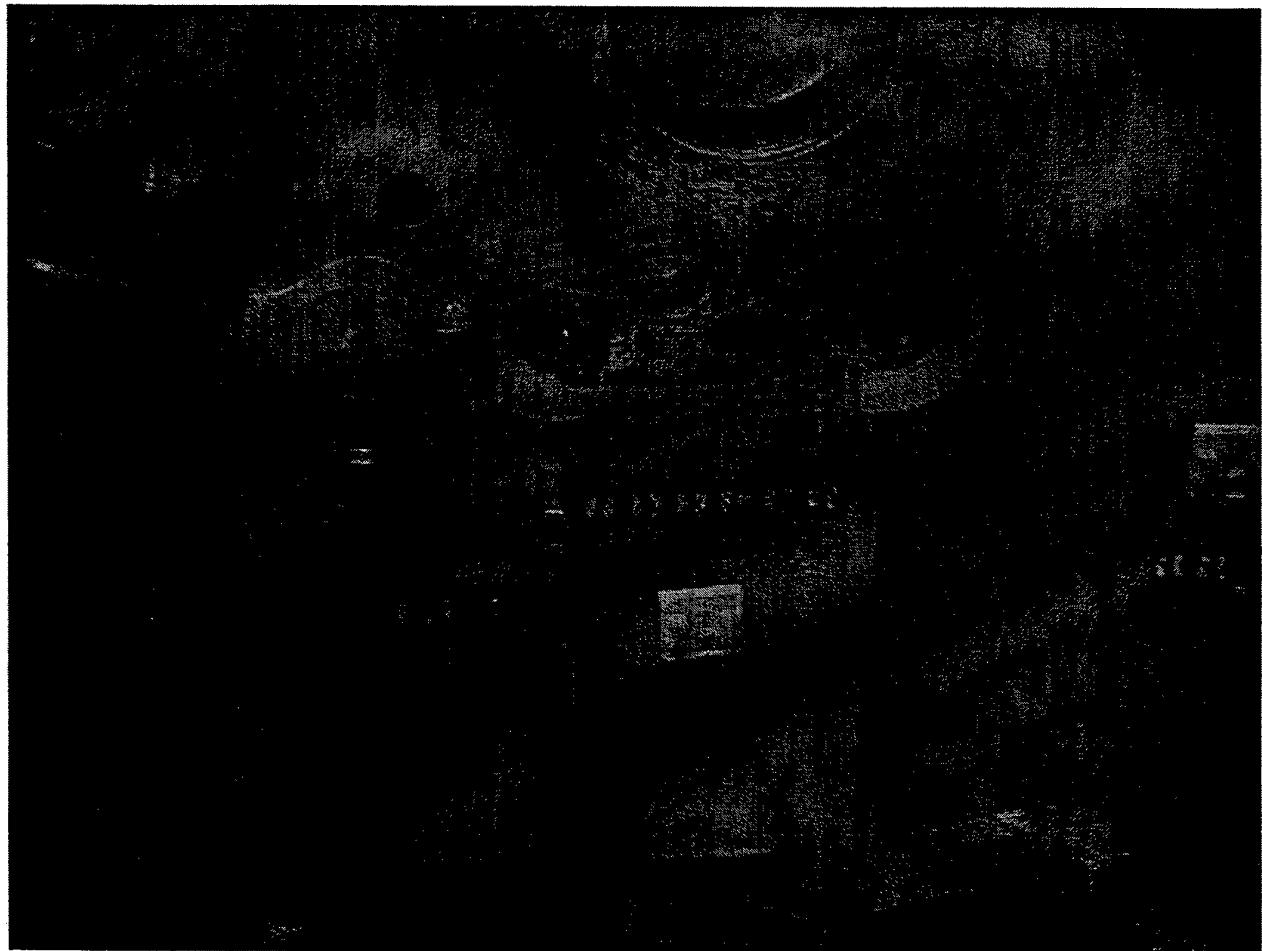
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## DAMAGE TOLERANCE – FRACTURE MECHANICS

The experience on SCT specimens at CAL confirmed that for single loading path elements, inspection intervals for cracks are very short if the crack is growing in stabilized flight conditions, were helicopter loading spectra are close to a CAL

In practice the requirement in this case is 'No Growth', with possibility of slow crack growth limited to start-stop or GAG cycles only.

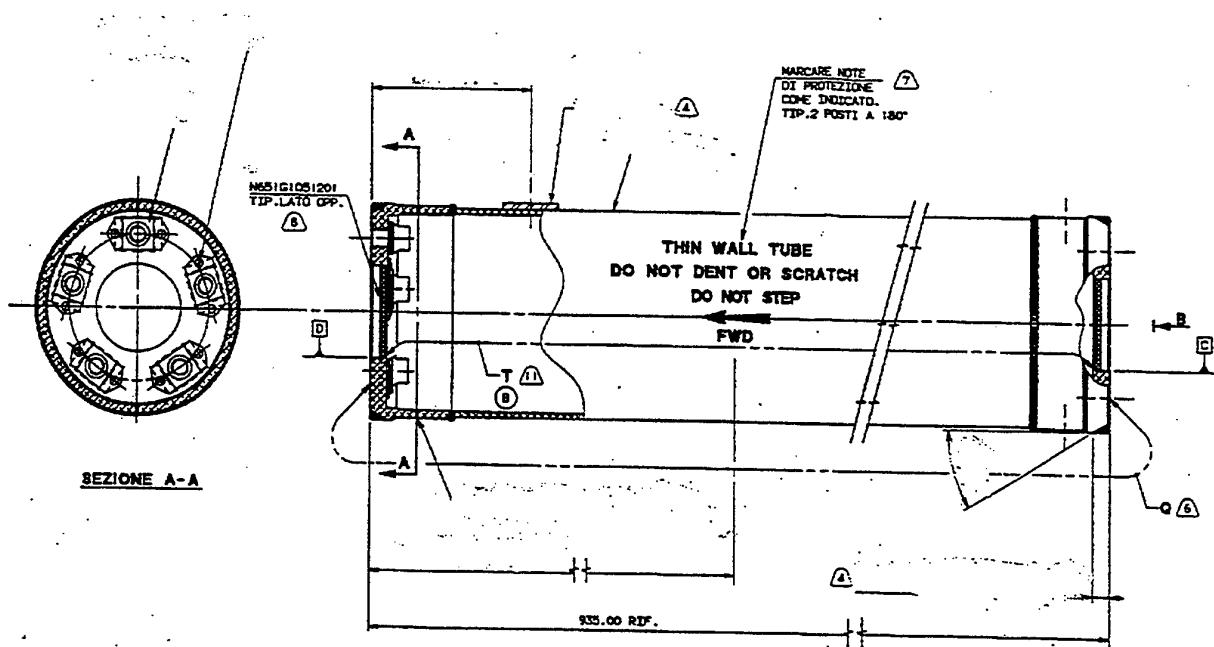
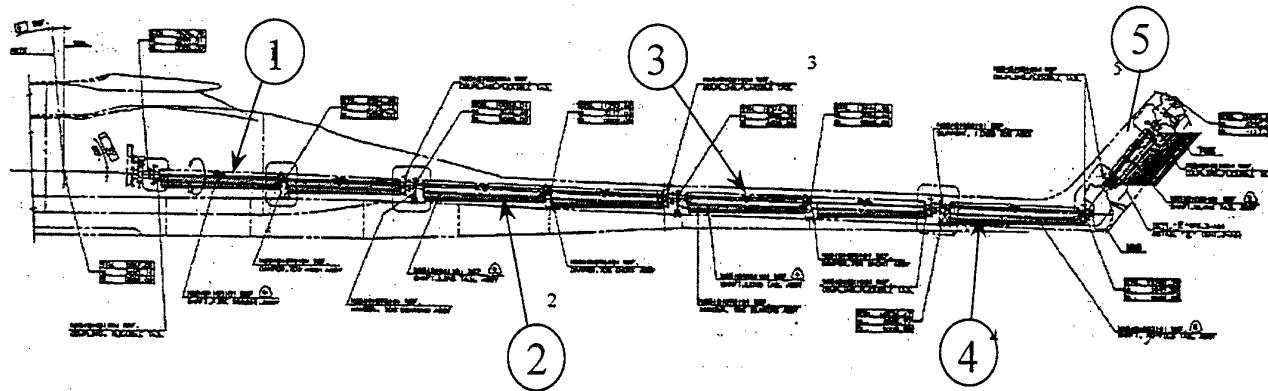
Crack sizes taken into account are based on MIL Spec:

|                |  |
|----------------|--|
| 0.38 mm radius | mechanical parts<br>Machined parts of the fuselage |
| 1.27 mm radius | panel & stringers of the fuselage                  |

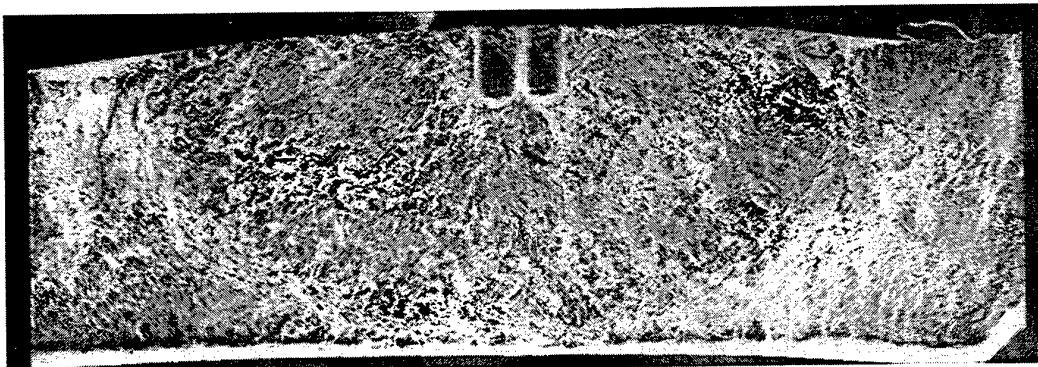
Two cases were carried out to address applicability:

- Tail Rotor Drive-line shafts as 'no growth or benign propagation'
- Inspections of panel / stringers of the naval rear fuselage of EH101 (500-1000 h)

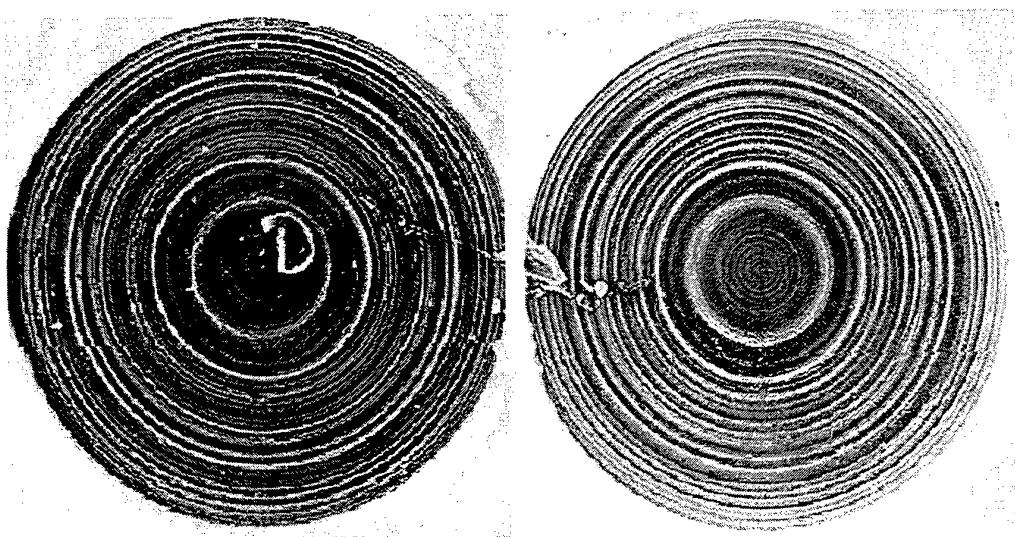
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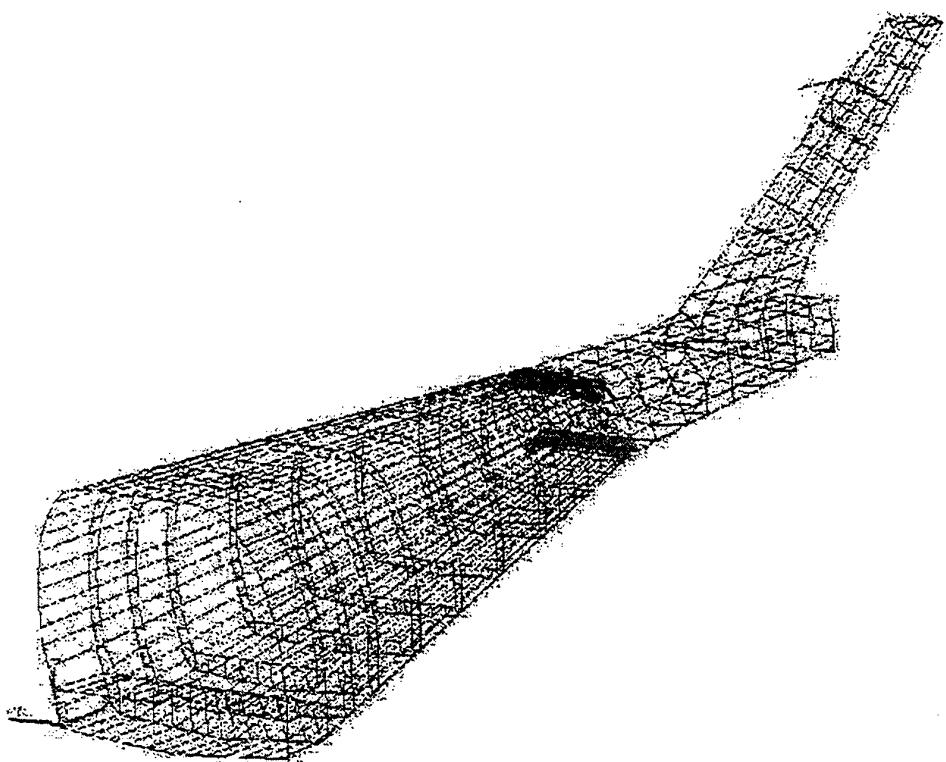
TR Transmission Shaft



Micro-notched specimens for fatigue tests - details of the failure section.



'No crack growth' test data at SEM : a) 'no growth' of the surface crack b) 'no growth' of the crack at the tip of the notch.



**US HELICOPTER INDUSTRY  
DAMAGE TOLERANCE ANALYSIS AND DESIGN**

**WORKSHOP ON DAMAGE TOLERANCE IN HELICOPTERS**

Cranfield University  
April 4-5, 2000

Presented by: George Schneider - Sikorsky Aircraft

Co-author: John Wang - Sikorsky Aircraft

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the U.S rotorcraft industry and the Government under RTTA/NASA cooperative  
agreement NCCW-0076, Advanced Technology, August, 1995

# US HELICOPTER INDUSTRY

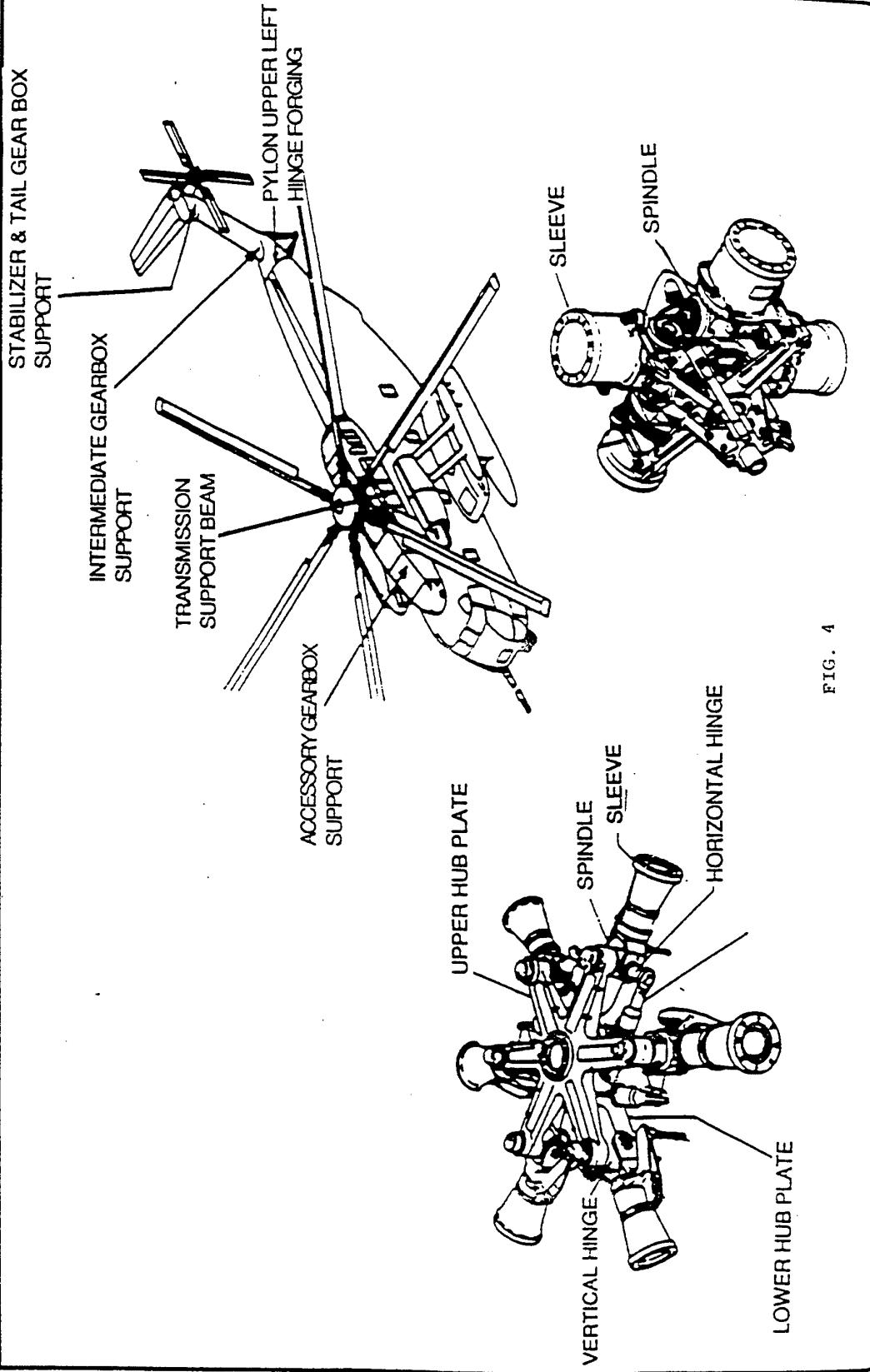
## DAMAGE TOLERANCE ANALYSIS AND DESIGN

### (METALLIC STRUCTURE)

- H-53 Helicopter Sikorsky / Airforce Damage Tolerance Project - 1980's
- Damage Tolerance and Flaw Tolerance Applied to Current Helicopters
  - Sikorsky      - Damage Tolerance Inspection Intervals - Current Helicopters
    - Flaw Tolerance Inspection Intervals - Current Helicopters
    - S-92 Rotor - FAR29.571 Certification, Amend. 28. October 1989
  - Bell            - Theoretical Redesign for Damage Tolerance - Current Helicopters
    - Bell 430 Rotor Damage Tolerance Certification
- RITA Damage Tolerance Project
  - Bell crack growth testing, crack growth code evaluations, NDI studies
  - Sikorsky crack growth models and equivalent initial flaw (crack) sizes.
  - FAA/RITA Interaction
- Conclusions and Key Issues

# AIR FORCE H-53 DAMAGE TOLERANCE ASSESSMENT

## STRUCTURE ANALYZED



## SOME POTENTIAL MAIN ROTOR CRACK LOCATIONS

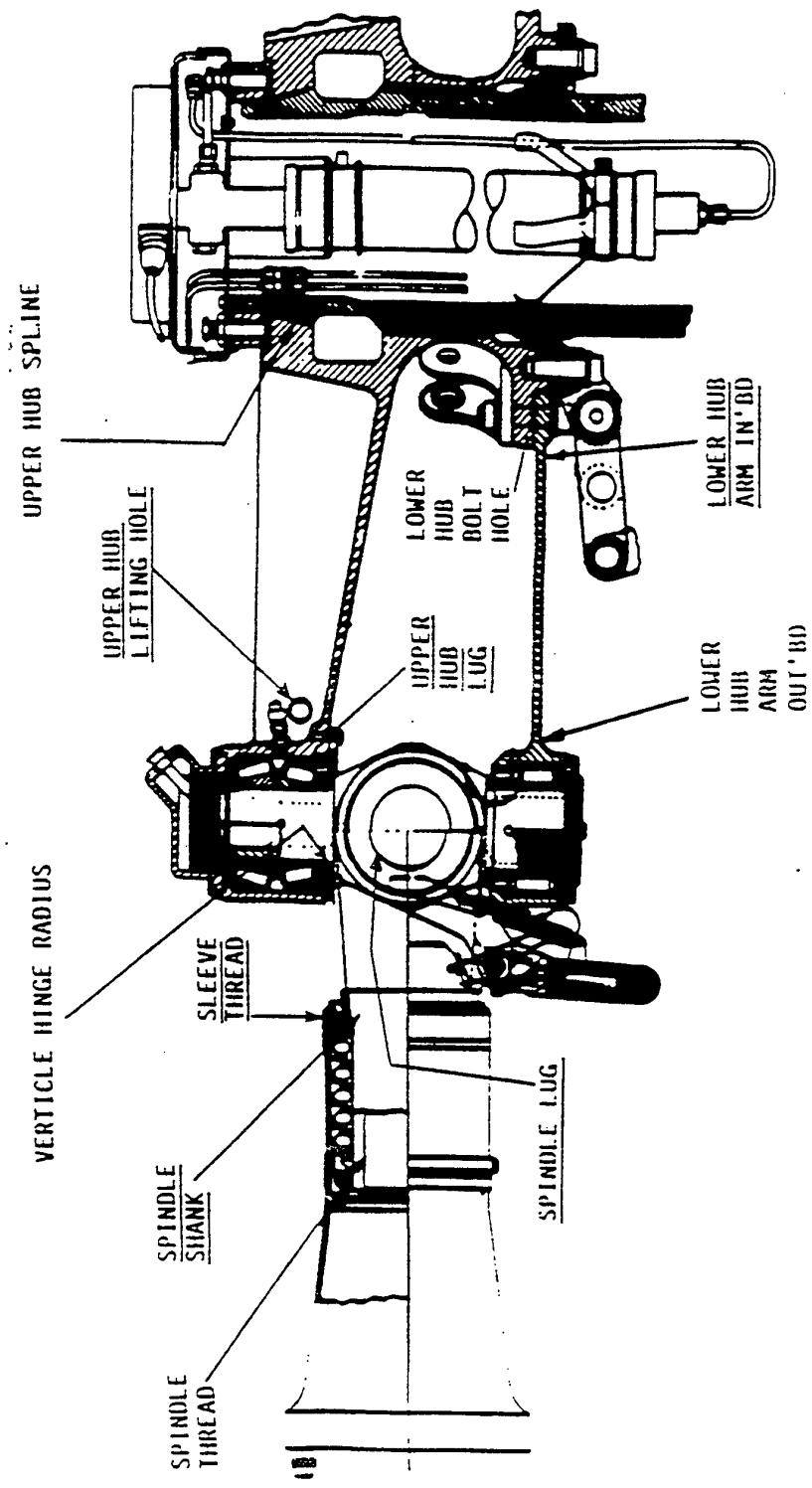


FIG. 5  
DAMAGE TOLERANCE ASSESSMENT PERFORMED FOR UNDERLINED LOCATIONS

## MAIN ROTOR CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)  
 Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

| Structure<br>- Crack Location       | Material                | High Cycle<br>Threshold<br>Crack Depth<br>(inches) | Crack Propagation Time*<br>(Flight Hours)<br>From Initial Crack Depth of:<br>0.005" 0.010" 0.030" |
|-------------------------------------|-------------------------|--|---|
| <i>H-53</i>                         |                         |  |   |
| Upper Hub Plate<br>- Lifting Hole   | Ti-6Al-4V<br>Alpha-Beta | 0.003<br>0.003+                                    | 380<br>High<br>50<br>16<br>5000+  |
| Lug                                 |                         |  |   |
| Lower Hub Plate<br>- Inboard Arm    | Ti-6Al-4V<br>Alpha-Beta | 0.008  | High<br>558<br>61   |
| Horizontal Hinge<br>- Damper Radius | 4340 Steel              | 0.003  | 212<br>14<br>--   |
| Spindle<br>- Lug                    | Ti-6Al-4V<br>Alpha-Beta | 0.022<br>0.012                                     | --<br>--<br>>2000<br>>2000<br>270<br>80   |
| Sleeve<br>- Threads                 | Ti-6Al-4V<br>Alpha-Beta | 0.002  | 150<br>42<br>10   |
| Blade Spar                          | Al 6061-T6              | 0.030  | --<br>>2000<br>271  |

FIG. 6

## TAIL ROTOR CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)  
Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

| Structure<br>- Crack Location | Material  | High Cycle<br>Threshold<br>Crack Depth<br>(inches) | Crack Propagation Time*<br>(Flight Hours)<br>From Initial Crack Depth of:<br>0.005" 0.010" 0.030" |
|-------------------------------|-----------|--|---|
| H-53                          | Ti-Al-4V  | 0.023<br>0.007                                     | 433<br>10<br>56<br>7<br>10<br>--  |
| Sleeve<br>- Threads           | Ti-6Al-4V | 0.002  | -->2000 475   |

FIG. 7

## AIRFRAME CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)  
Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

| Structure<br>- Crack Location                                  | Material                | High Cycle<br>Threshold<br>Crack Depth<br>(inches) | Crack Propagation Time*<br>(Flight Hours)<br>From Initial Crack Depth:<br>0.005" 0.010" 0.030" |
|--|-------------------------|--|--|
| H-53   |                         |  |  |
| Upper Left Pylon Al7075-T73<br>Fold Hinge Ftg.<br>- Rivet Hole | 0.005                   | 410  | 170 45   |
| Acc. Gear Box Al7075-T73<br>Support Fitting<br>- Bolt Hole     | 0.002                   | 50   | 29 7   |
| Stabilizer Support Al7075-T73<br>Fitting<br>- Hole             | 0.004                   | 40   | 21 6   |
| Transmission Al7075-T73<br>Support Frame<br>- Holes            | 0.050<br>0.100<br>1.000 | ---<br>---<br>---                                  | >2000<br>>2000<br>>2000  |

FIG. 8

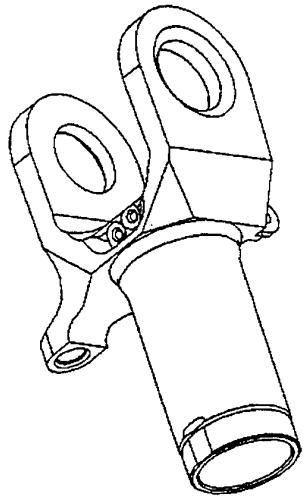
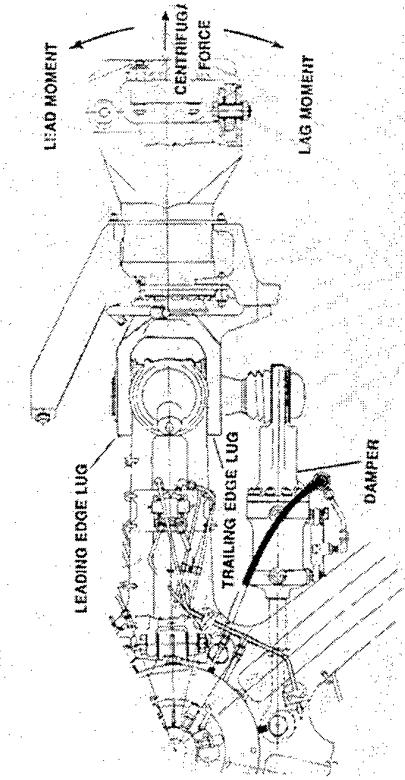
# SIKORSKY- DAMAGE TOLERANCE INSPECTION INTERVALS

## Examples:

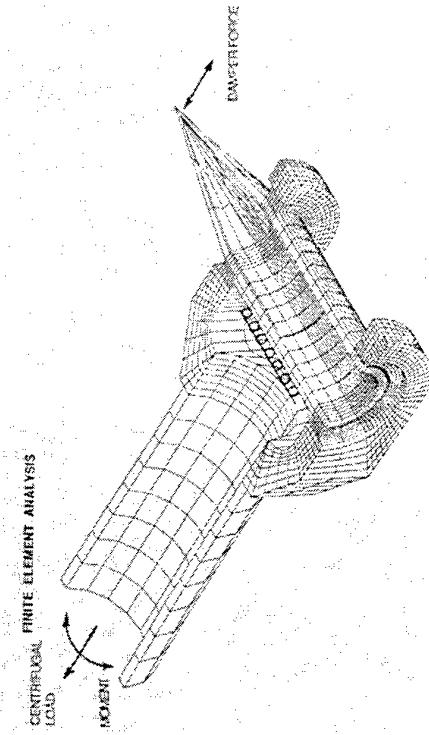
| <u>Part</u>            | <u>Material</u> | <u>Inspection Method</u> | <u>Detectable Crack Size (inches)</u> | <u>Inspection Interval (Flt Hours)</u> |
|------------------------|-----------------|--------------------------|---------------------------------------|--|
| • S-61 Spindle Shank   | 4340 steel      | UT                       | 0.020                                 | 50                                     |
| • H-53E Spindle Lug    | Ti-6Al-4V       | UT                       | 0.080                                 | 50                                     |
| • H-60 Hub Damper Hole | Ti-6al-4V       | FP, EC                   | 0.080                                 | 200                                    |

# H-53E SPINDLE LUG CRACK GROWTH INSPECTION INTERVAL

Lug crack under bonded liner resulting from fretting



## Finite Element Model



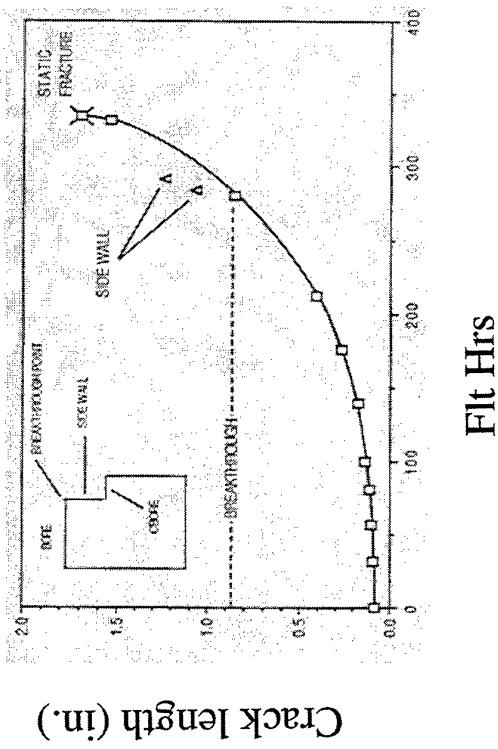
# H-53E SPINDLE LUG CRACK GROWTH INSPECTION INTERVAL

Lug crack under bonded liner resulting from fretting

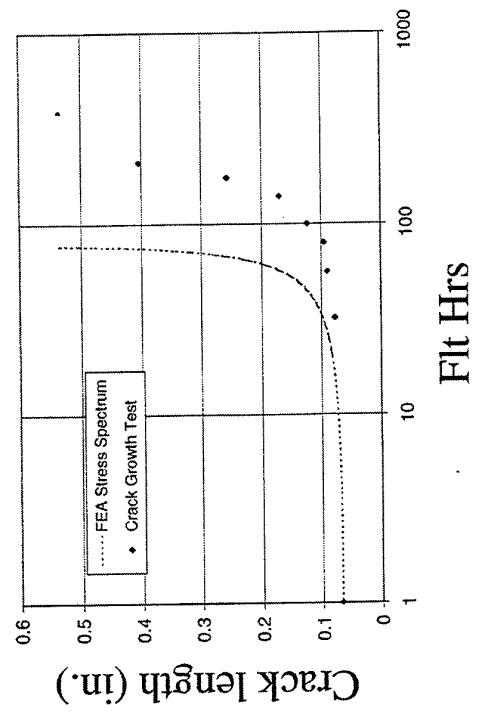
Analysis: NASGRO strip yield corner crack model

Finite element analysis stress distribution

## Spectrum Test Results



## Spectrum Analysis Results



Ultrasonic inspection for 0.08 inch crack  
Inspection interval 50 flight hours

# SIKORSKY - FLAW TOLERANCE INSPECTION INTERVALS

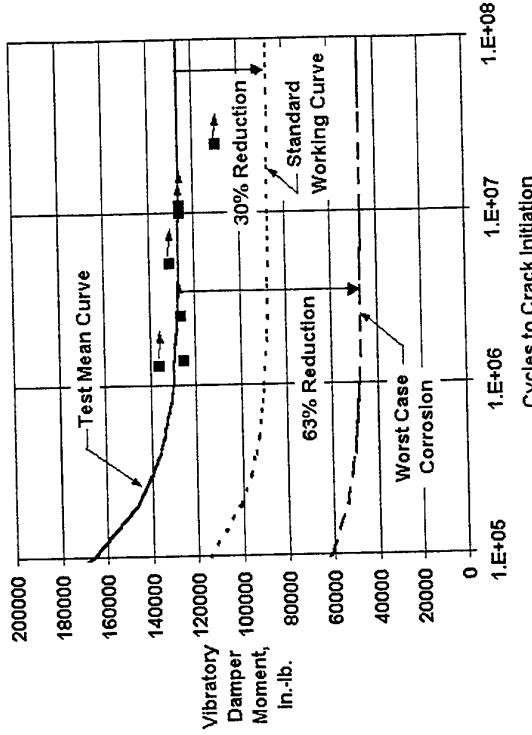
## Examples:

| <u>Part</u>                 | <u>Material</u> | <u>Service Degradation</u> | <u>Inspection Interval (Flt Hrs)</u> |
|-----------------------------|-----------------|----------------------------|--------------------------------------|
| • CH-53A/D Horiz. Hinge Pin | 4340 steel      | corrosion                  | 1500 (1200)                          |
| • S-76 TR Pitch Horn        | Aluminum        | corrosion                  | 12000 *                              |

\* Replacement time set to this interval

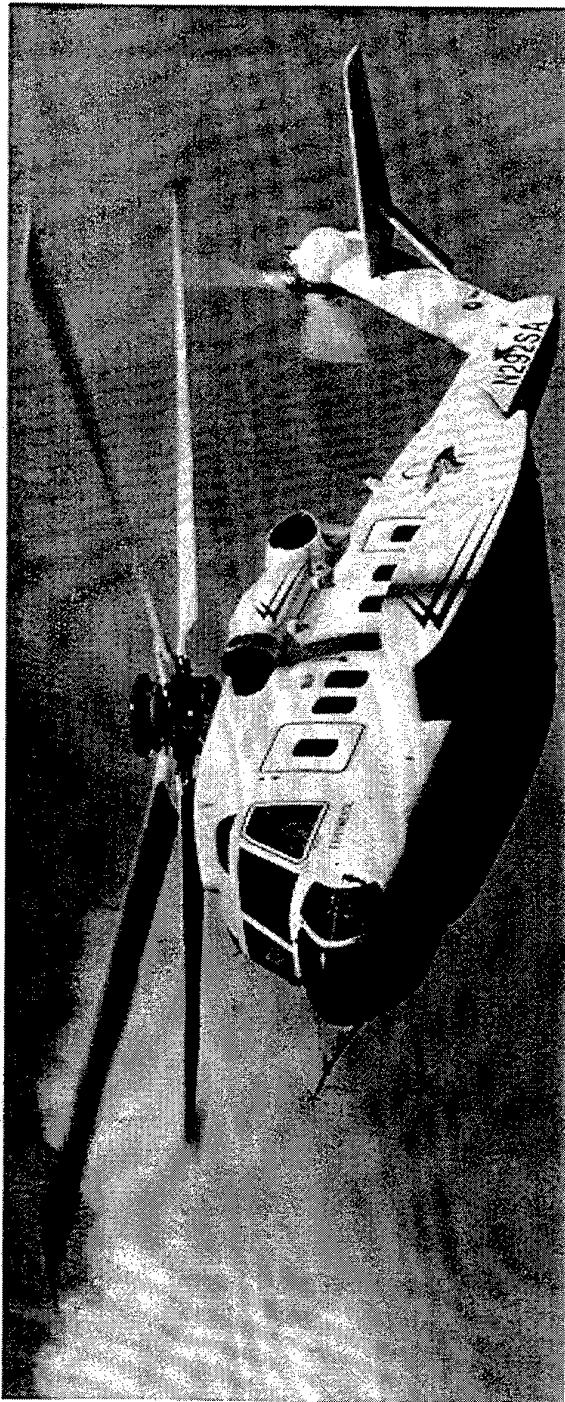
# CH-53 A/D HORIZONTAL HINGE PIN FLAW TOLERANT APPROACH - INSPECTION INTERVAL

- Subject to in service corrosion
- Coupon testing: “worst case” 63 % reduction
- Reduction applied for full scale test data
- Miner’s Analysis => 1500 flt hours
- Inspection Interval 1200 flt hours
- Replacement Time 8300 Flt Hours



# SIKORSKY S-92 - FAR 29.571 ROTOR CERTIFICATION

## METAL ROTOR COMPONENTS FLAW TOLERANT SAFE-LIFE EVALUATION



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Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

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# SIKORSKY S-92 - FAR 29.571 ROTOR CERTIFICATION

## METAL ROTOR COMPONENTS FLAW TOLERANT SAFE-LIFE EVALUATION

### FAA Requirements:

- Replacement Time: based on barely visible flaws
- Inspection Interval: based on severe flaws

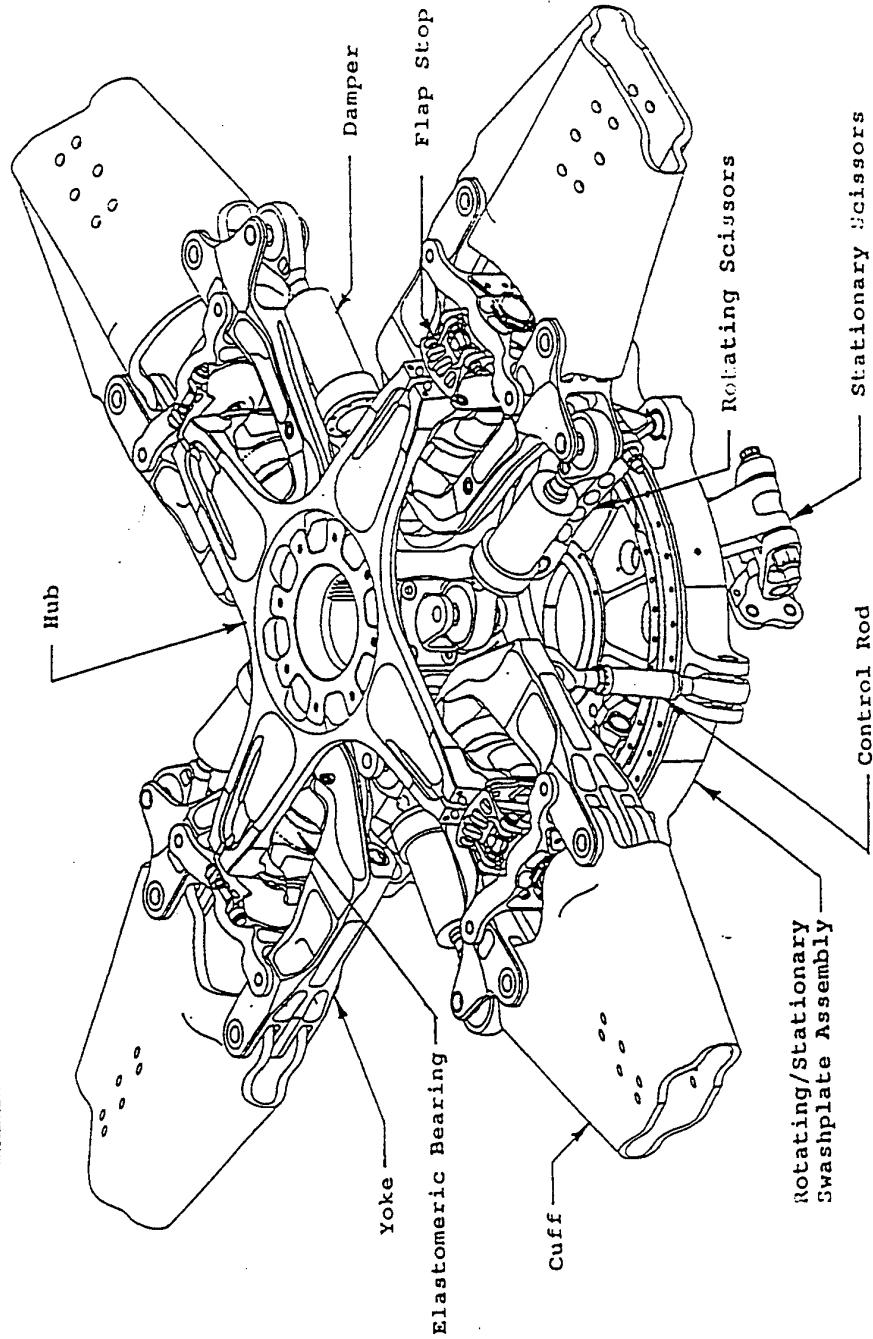
Flaw types and sizes based on SAC review of field damage reports

Coupon tests to evaluate effects of flaws

Full scale tests performed with barely visible flaws

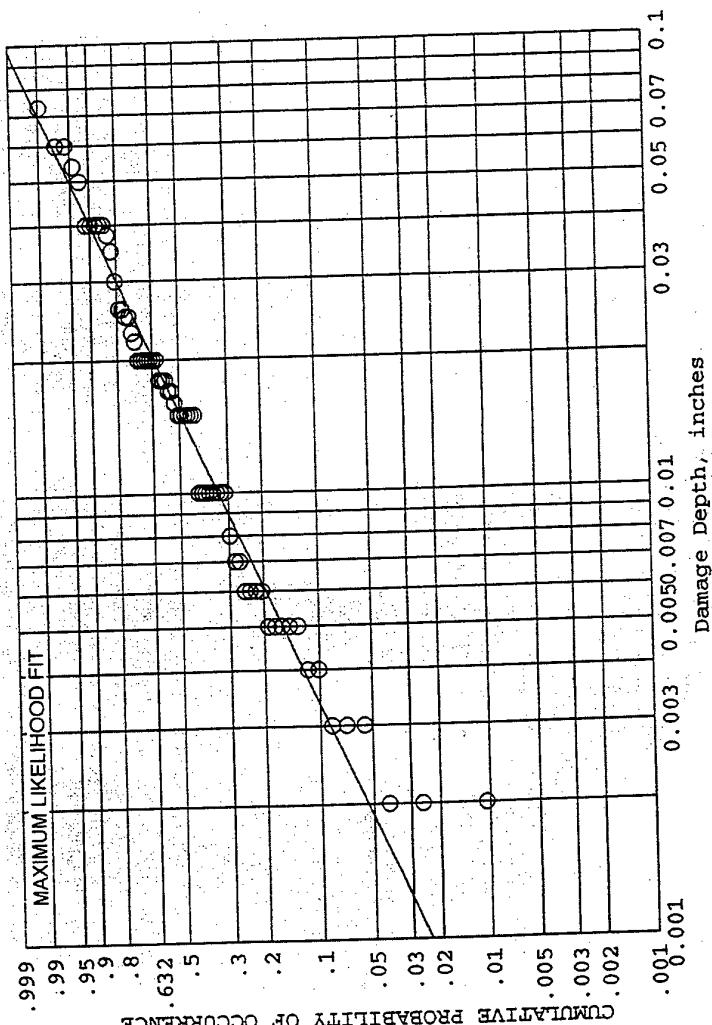
Miner's analysis used for determining replacement times

# SIKORSKY- S-92 ROTOR HEAD



# SIKORSKY HISTORY OF SERVICE-RELATED DAMAGE

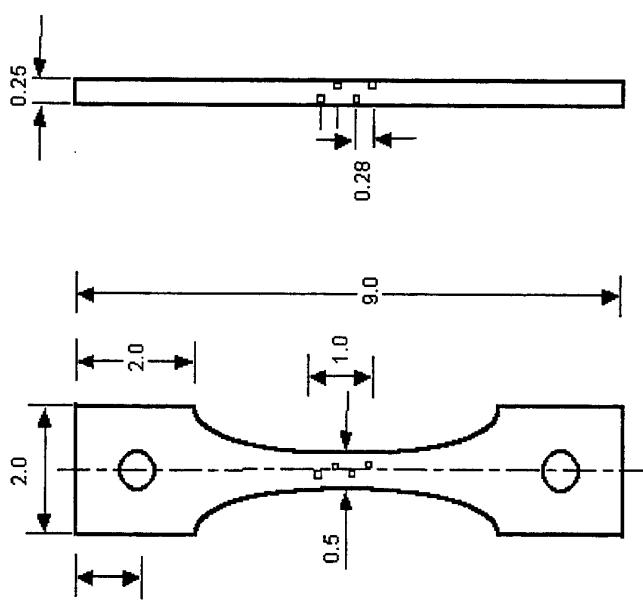
66 S-76 Aluminum and Titanium Rotor Head Parts  
53 Parts Damage > 0.005 inch depth  
8000 S-76 Rotor Head Parts in Service



Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

# FLAWED SPECIMEN TESTING

AI7075-T73, Tested at 0 Steady Stress



□ Defect introduced into test specimen  
Gage Section (not to scale)

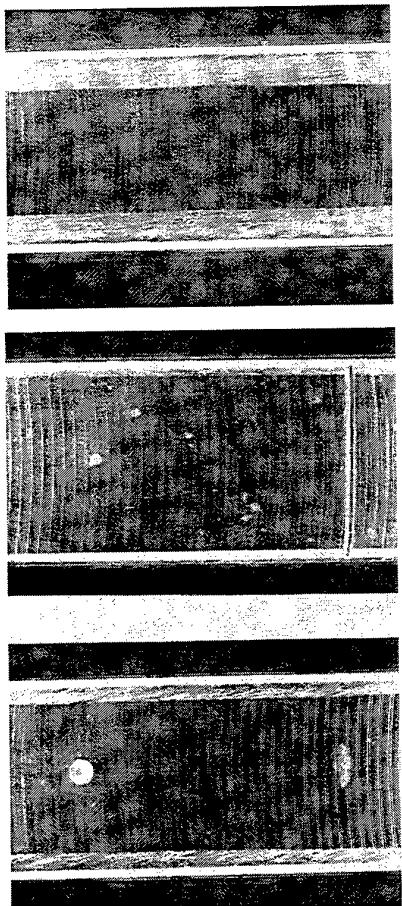
All Dimensions in Inches

| Surface                     | Flaw Type  | Depth Range                               |
|-----------------------------|--|---|
| Highly Polished<br>Polished | N/A<br>N/A   | N/A<br>N/A                                |
| As-Machined                 | N/A  | 0.00012                                   |
| Lightly Polished            | N/A  | 0.00005                                   |
| 0.005"Deep Natural Flaws    | Corrosion Pit<br>Sharp Scratch<br>Gouge<br>Sharp Indent<br>Dull Indent | .0050-.0068                               |
| 0.005"Deep Artificial Flaws | EDM Notch<br>Milled Notch<br>ECM Pit                                   | .0040-.0054<br>.0045-.0054<br>.0040-.0064 |
| 0.040"Deep Natural Flaws    | Corrosion Pit<br>Sharp Scratch<br>Gouge<br>Sharp Indent<br>Dull Indent | .034-.045                                 |
| 0.040"Deep Artificial Flaws | EDM Notch<br>Milled Notch<br>ECM Pit                                   | .0040 Nom<br>.039-.043<br>.038-.042       |
| 0.075"Deep Natural Flaws    | Gouges   | 0.071-0.080                               |

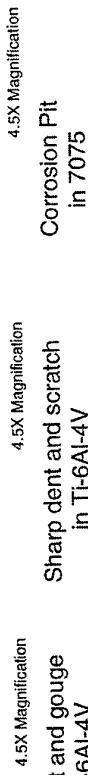
RITA

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# TYPICAL 0.005 INCH DEEPEP FLAWS



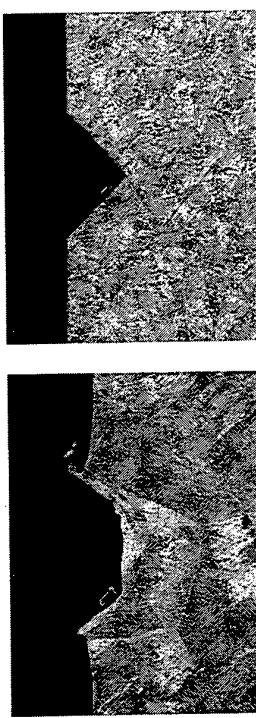
4.5X Magnification  
Dull dent and gouge  
in Ti-6Al-4V



4.5X Magnification  
Sharp dent and scratch  
in Ti-6Al-4V



4.5X Magnification  
Corrosion Pit  
in 7075

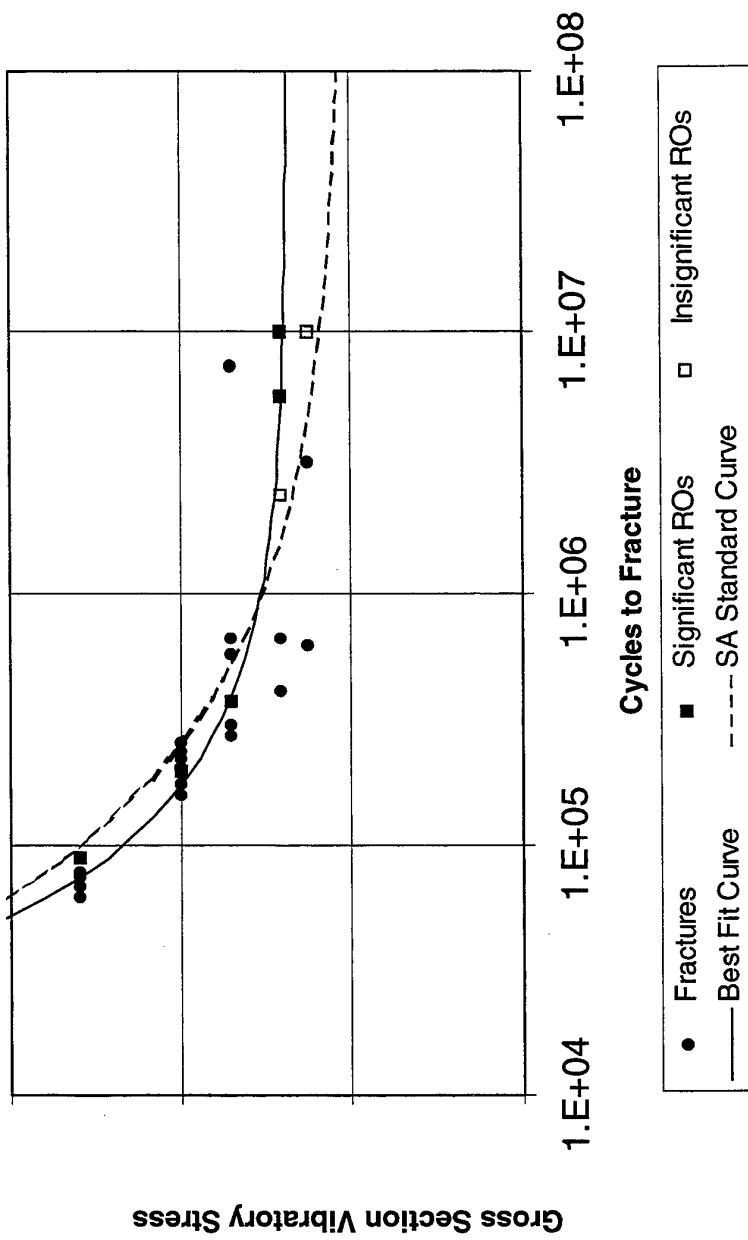


100X Magnification  
Scratch  
in Ti-6Al-4V

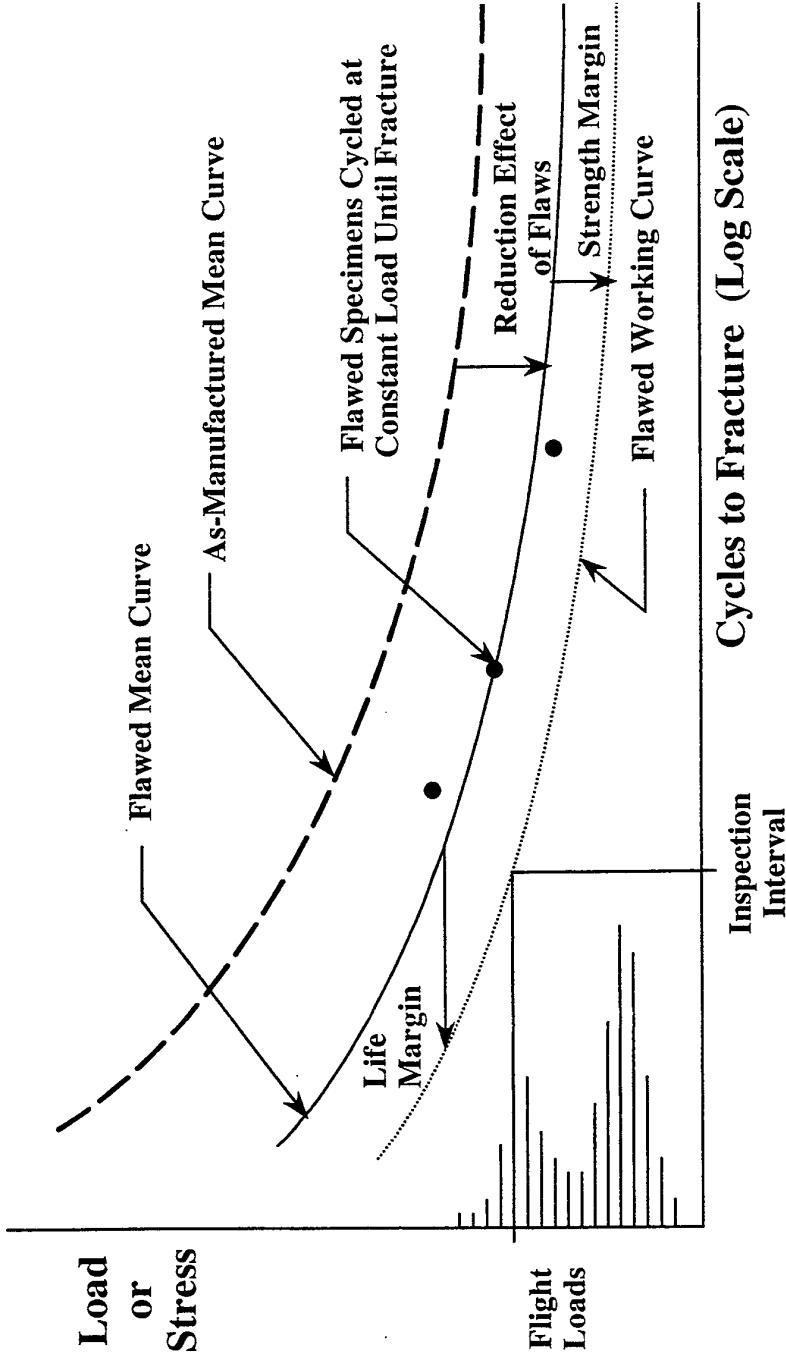


100X Magnification  
Sharp milled notch  
in Ti-6Al-4V

**AL7075-T73 -0.005 INCH CORROSION**



# FULL SCALE FLAW TOLERANCE TEST



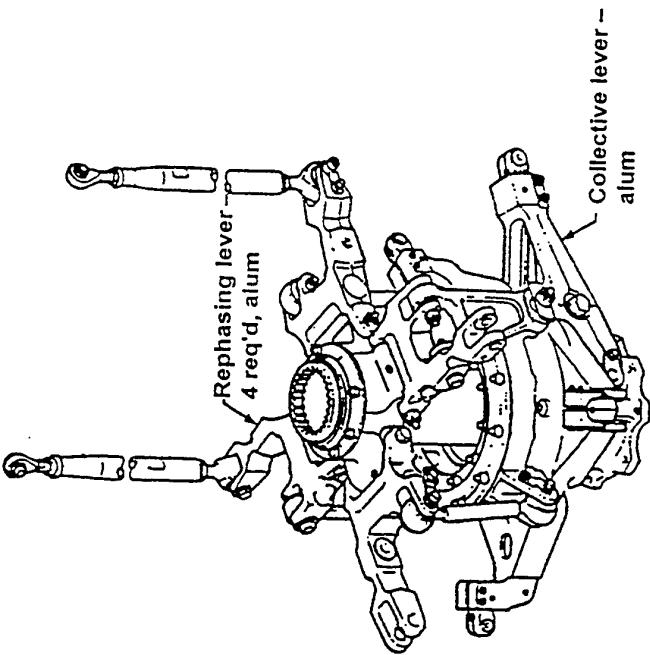
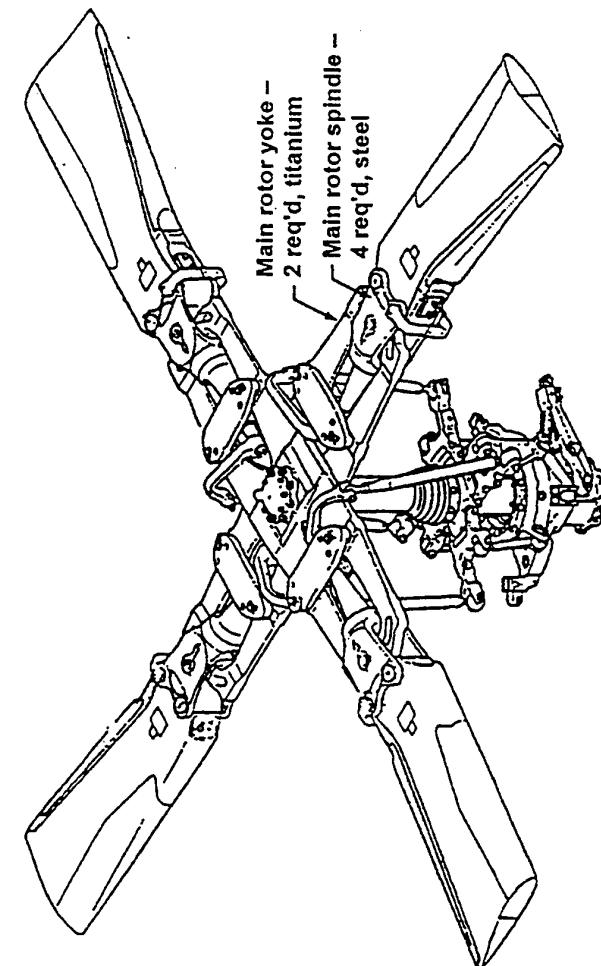
Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

# BELL - THEORETICAL REDESIGN FOR DAMAGE TOL.

FAA Investigation - Four selected rotor parts

Presented at 55th Annual AHS Forum, May 1999

James Michael, Bill Dickson, Tom Campbell



# BELL - THEORETICAL REDESIGN FOR DAMAGE TOL.

FAA Investigation - Four selected rotor parts

Presented at 55th Annual AHS Forum, May 1999  
James Michael, Bill Dickson, Tom Campbell

Approach:

No Growth

Slow Growth = Twice replacement time

Initial Crack Size - 0.015 inch deep (0.381 mm) semicircular

Four Parts:

Main Rotor Yoke

Main rotor Spindle

Main rotor Rephase Lever

Collective Lever

- Ti-6Al-4V beta stoa

- 15-5 Stainless steel

- Al7075-T73

- Al7075-T73

# BELL - THEORETICAL REDESIGN FOR DAMAGE TOL. MAIN ROTOR SPINDLE

15-5 Stainless Steel

$$\Delta K_{th0} = 5.0 \text{ ksi(in)}^{1/2}$$

Safe Life Repl. Time = 10000 flt hrs

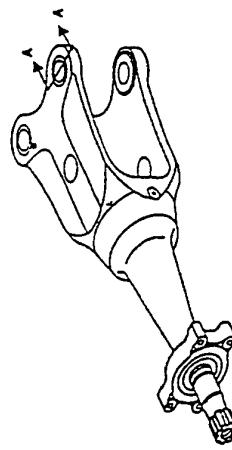
Crack Growth Time = 143 flt hrs

Original Thickness = 0.69 in

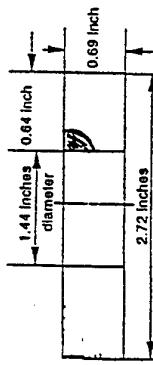
Revised Thickness = 0.95 in. ( 38 %) (or 1.14 inches)  
(no crack growth)

Crack Growth Time = 20,000 flt hrs

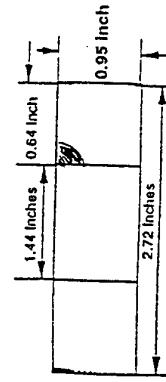
Weight increase (2 lugs) = 1.13 lbs. (4.6%) per aircraft



Main Rotor Spindle



Section A-A



Redesign - Section A-A

# BELL - THEORETICAL REDESIGN FOR DAMAGE TOL. SUMMARY

|                    | Baseline<br>safe-life<br>retirement<br>time | Baseline<br>damage tolerant<br>crack growth life<br>(Ref. 2) | Calculated damage<br>tolerant crack growth<br>life <sup>(a)</sup> | Calculated<br>damage tolerant in-<br>spection interval <sup>(a)</sup> | Material                       | Weight increase<br>over baseline<br>assembly (%) |
|--------------------|---|--|---|---|--------------------------------|--|
| Helicopter PSE     | 5000 hrs                                    | 20 hrs   | N/A <sup>(b)</sup>  | N/A <sup>(b)</sup>  | 6AL-4V titanium with<br>BSHTOA | N/A <sup>(b)</sup>                               |
| Main rotor yoke    | 5000 hrs                                    | 143 hrs  | > 20,000 hrs  | 10,000 hrs  | 15-5 stainless<br>steel        | 4.6%   |
| Main rotor spindle | 10,000 hrs                                  | 13 hrs   | No crack growth   | No inspection re-<br>quired   | 7075-T73 aluminum              | 22%  |
| Collective lever   | 10,000 hrs                                  | 13 hrs   | No crack growth   | No inspection re-<br>quired   | 7075-T73 aluminum              | 15%  |
| Rephase lever      | 5000 hrs                                    | 78 hrs   | No crack growth   | No inspection re-<br>quired   | 7075-T73 aluminum              | 15%  |

(a) Crack growth life based on limited analytical study results for theoretical designs of each PSE.  
 (b) Composite material might be an alternative for the main rotor yoke to potentially meet damage tolerance requirements with a 5,000-hour inspection interval.

## BELL - 430 ROTOR CERTIFICATION



Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

## BELL - 430 ROTOR CERTIFICATION

Transport Canada required certification to FAR29.571 Amendment 28

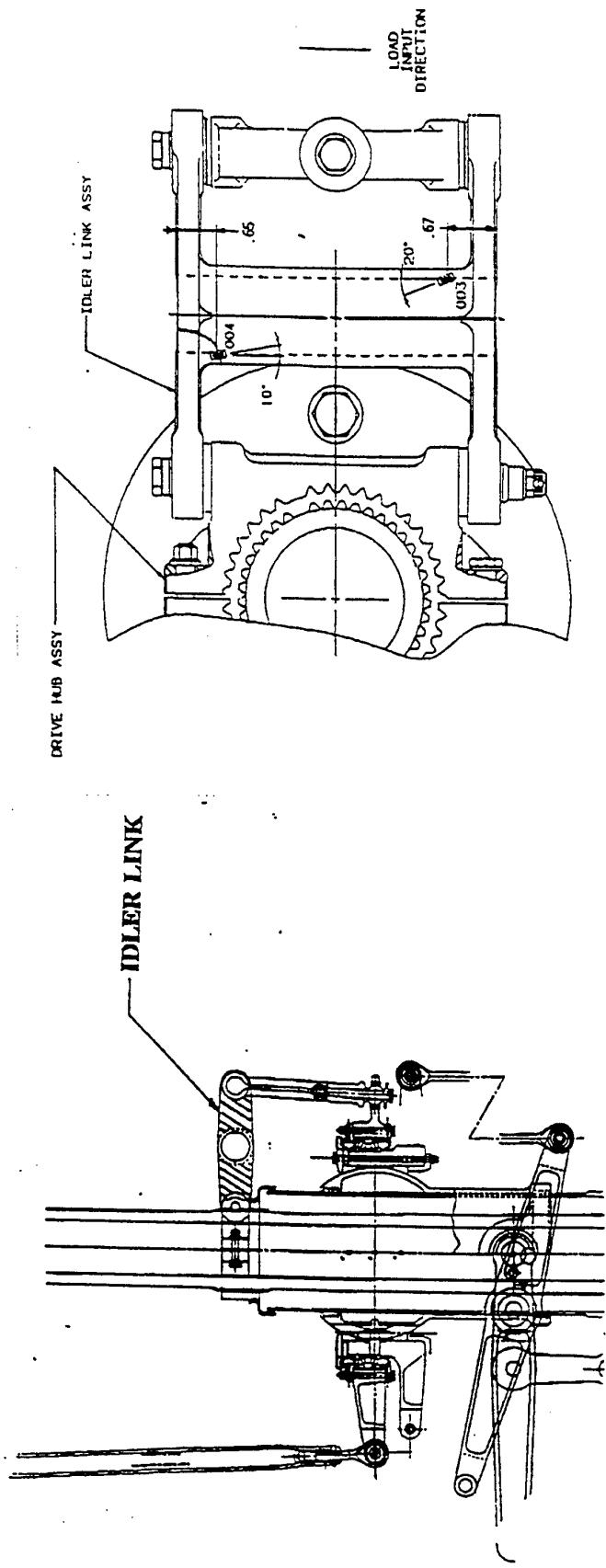
- Bell Criteria - No growth or two life times from a 0.015 inch deep crack
  - Used similar design criteria for more than 10 years

Full scale S-N testing with strain surveys to validate analytical models

|                        |  |
|------------------------|--|
| Crack growth analysis: | 0.015 inch deep crack will not grow, or<br>two lifetimes from an 0.015 inch deep crack |
|------------------------|--|

# BELL - 430 ROTOR CERTIFICATION

## SWASHPLATE DRIVE ASSEMBLY IDLER LINK



## **BELL - 430 ROTOR CERTIFICATION**

### **SWASHPLATE DRIVE ASSEMBLY IDLER LINK**

- Strain measured at crack origin location
- Measured strain above crack growth threshold.
- Part redesigned to reduce strain by a minimum of 15 %.
- Subsequent test of new design - Local strain below crack growth threshold

# **Rotorcraft Industries Association (RITA)**

## **Damage Tolerance For Helicopter Structure**

**Jim Cronkhite**  
**Bell Helicopter Textron Inc.**

**George Schneider**  
**Sikorsky Aircraft**

Technical tasks in this document include tasks supported with shared funding by  
the U.S rotorcraft industry and the Government under RITA/NASA cooperative  
agreement NCCW-0076, Advanced Technology, August, 1995

Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

**RITA**

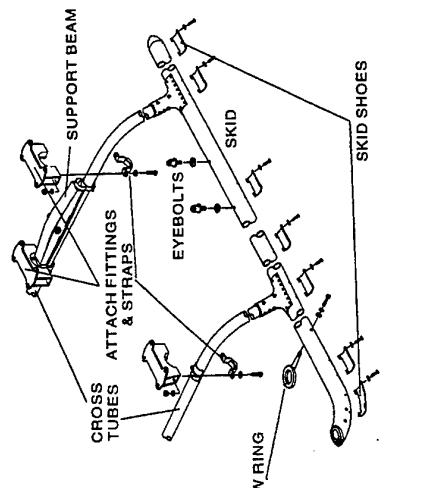
## **Bell Helicopter Textron Inc.**

**Global Project Objective:** Improve safety and reduce design and O&S costs through improvements in damage tolerance methods and tools.

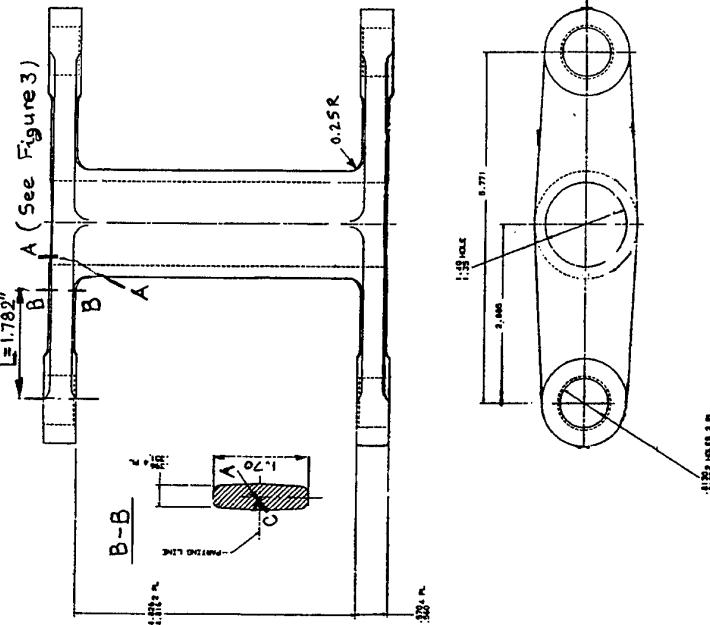
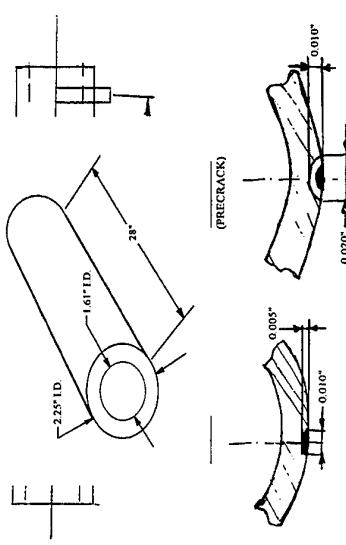
- Software Validation (CRACKS98, NASGRO 3.0, etc.)
- Crack Growth Rate Data (C(T) and K<sub>b</sub>- bar specimens)
- Component Crack Growth Tests
  - Landing Gear Cross Tube
  - Idler Link
- NDE
  - Multilayer Aluminum Sheet Corrosion
  - Evaluate Operator Field Techniques
- Structural Integrity Methodology

# COMPONENT CRACK GROWTH TESTING

## M407 Skid Gear Attachment



SPECIMEN - GROMM



## M430 Idler Link Testing

RITA

Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

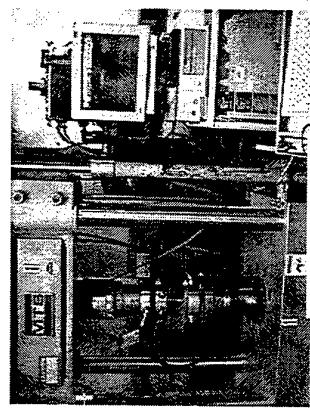


**General Objective:** Develop technical foundation for the transition from safe-life to damage tolerant design, certification, and management of helicopter structure.

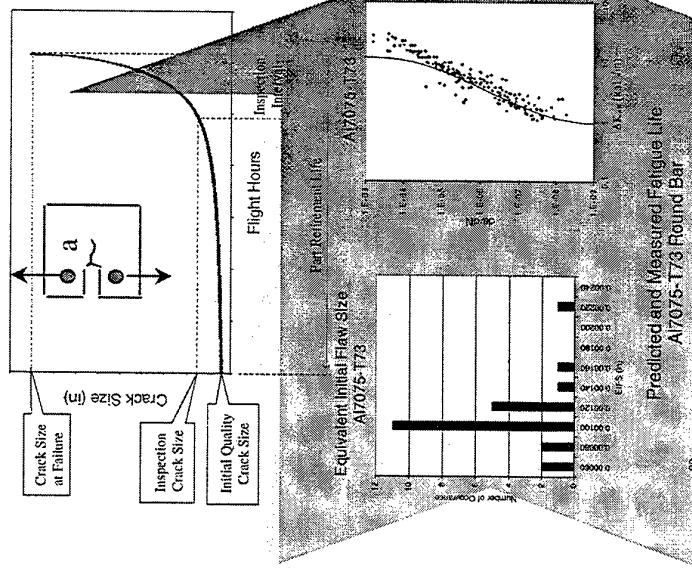
- FASTRAN, NASGRO 3.0, Crack Closure Evaluation
- Polished Specimen Fatigue and Crack Growth Tests
- NASGRO Strip Yield (Closure) Model Developed From Polished Specimen Data
- Machined and Flawed Specimen Fatigue Tests
- Crack Growth Analysis With Residual Stress - Shot Peening
- Corrosion / Crack Growth Testing
- Spectrum Load Polished, Machined, and Flawed Specimen Fatigue Tests
- Equivalent Initial Flaw Sizes (EIFS) - Best Fit and Distributions
- Full Scale Part Fatigue and Crack Growth Evaluation
- Final Report and Damage Tolerance Procedures Manual

# Damage Tolerance For Helicopter Structures

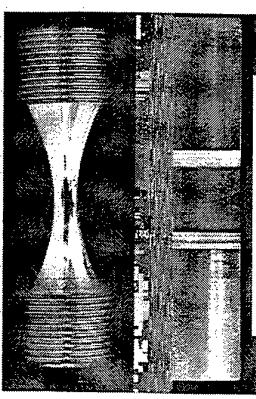
## Material Fatigue & Crack Growth Tests



## Damage Tolerance Crack Growth Model



## Test Setup



## Test Specimen



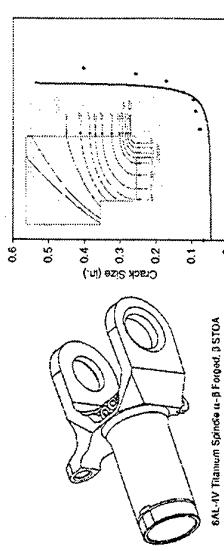
## Material Inclusion

**RITA**  
Microstructure Study  
Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000

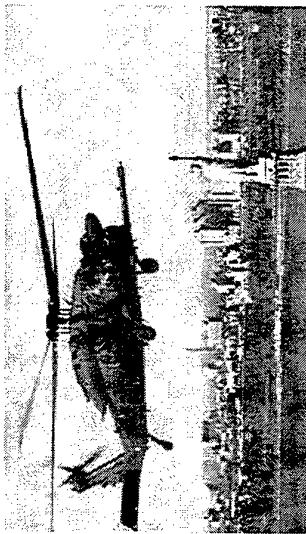
## Technology Application



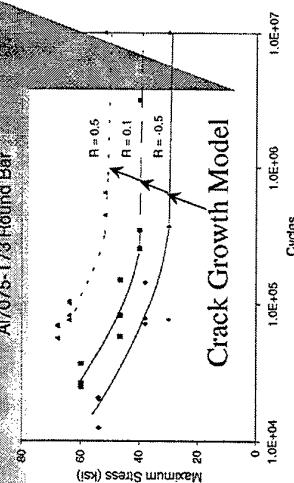
H-53E



## Component DTI Evaluation



**H-60 Black Hawk**  
Prediction vs. Measurement  
Workshop on Damage Tolerance in Helicopters, Cranfield University, April 2000



# CONCLUSIONS

- Damage tolerance requires stress reduction over safe-life.
- Bell theoretical study - 5% to 22 % weight increase (maybe less with optimization)
- one component switch to composites recommended
- Sikorsky H-53 study - rotor head redesign required for DT

## SOME ISSUES

- Crack Growth Rate and Threshold
  - better data base
  - sensitivity of design
  - standardization
- Threshold - problems with load shedding tests and current threshold data
- Life Enhancement -analysis methods needed
  - reluctance to shot peen and cold work due to cost
- NDE
  - 0.010 to 0.030 inch cracks for dynamic systems
  - economics and practicality
  - service inspection - cracks or flaws
- Certification testing requirements ?

## **SESSION B**

# **DAMAGE TOLERANCE IN DYNAMIC COMPONENTS**



*Cranfield*  
UNIVERSITY



**DERA**



# **U.S. AIR FORCE VIEWS ON DAMAGE TOLERANCE FOR HELICOPTERS**

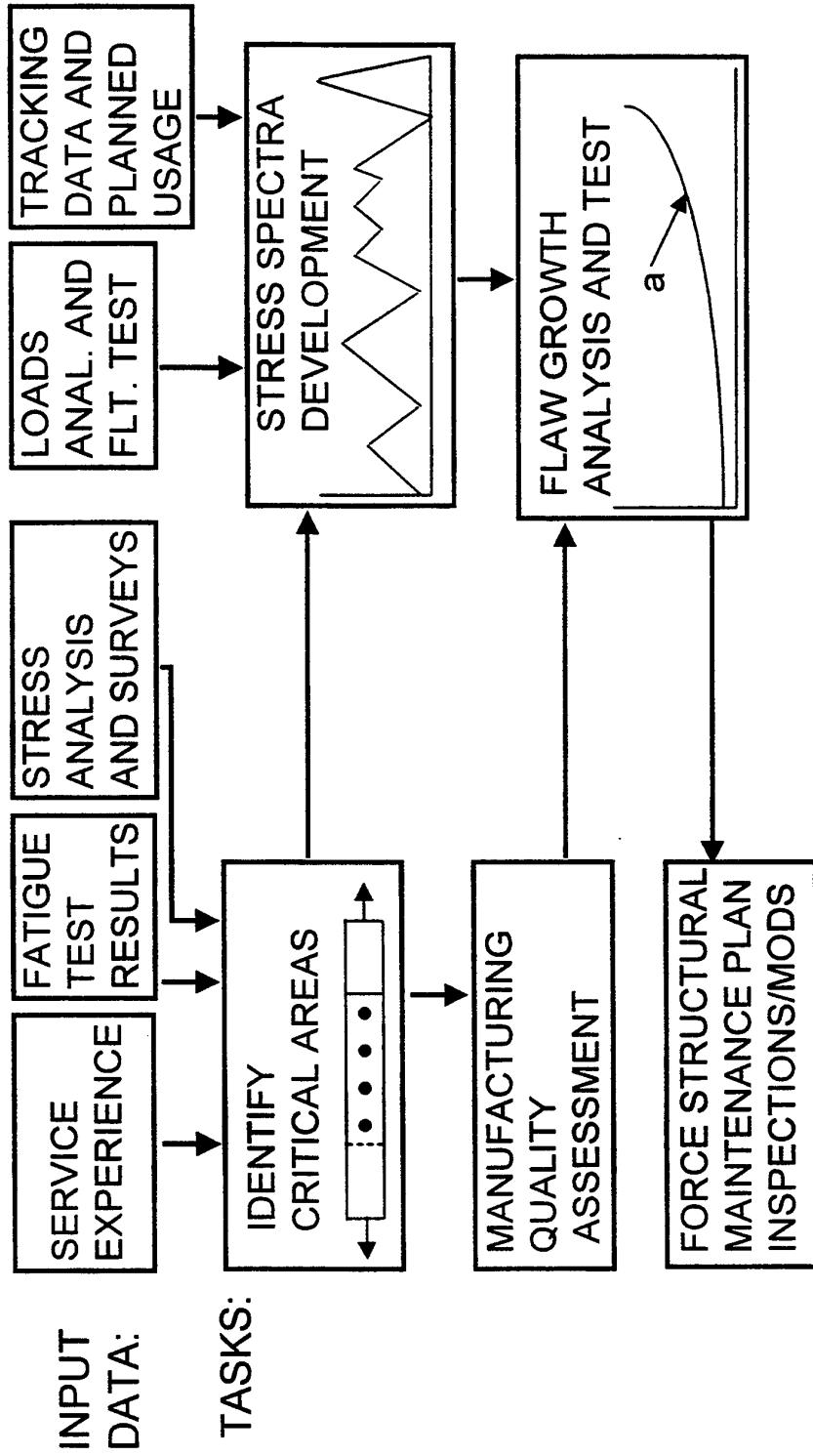
John W. Lincoln  
Engineering Directorate  
Aeronautical Systems Center  
Wright-Patterson Air Force Base, Ohio

## **BACKGROUND**

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- Success with damage tolerance assessments of fixed-wing aircraft and engines motivated the USAF to perform a damage tolerance assessment of a helicopter
- The USAF initiated a damage tolerance assessment on the Sikorsky HH-53 helicopter starting in 1983

# DAMAGE TOLERANCE APPROACH



OUTPUT: INSPECTION AND MOD REQUIREMENTS BY TAIL NUMBER

DAMAGE TOLERANCE EXPERIENCE - AIRCRAFT

```

graph TD
    Root[71,72,73,74,75,76,77,78,79,80,81,82,83,84,85,86,87,88,89] --> F4CD[F-4 C/D]
    Root --> C5[C-5]
    Root --> B1A[B-1A]
    Root --> F15[F-15]
    F4CD --> B52[B-52]
    C5 --> F4LES[F-4 (LES)]
    B1A --> C141A[C-141A]
    B1A --> C141B[C-141B]
    F15 --> F111[F-111]
    F4LES --> F5EF[F-5 E/F]
    F4LES --> T38[T-38]
    F5EF --> B1B[B-1B]
    T38 --> LEARJET[LEARJET]
    F5EF --> KC135[KC-135]
    T38 --> KC135SM[KC-135 SM]
    KC135 --> T46[T-46]
    E3A[E-3A]
    A7D[A-7D]
    A7D --> T37[T-37]
    C5SIEP[C-5 SIEP]
    SR71[SR-71]
    KC10A[KC-10A]
    HH53C[HH-53C]
    C141A --> C130[C-130]
    C141B --> C130
    C141B --> C17[C-17]
    KC135SM --> C130SM[C-130 SM]
    KC135SM --> C17
    KC10A --> A10[A-10]

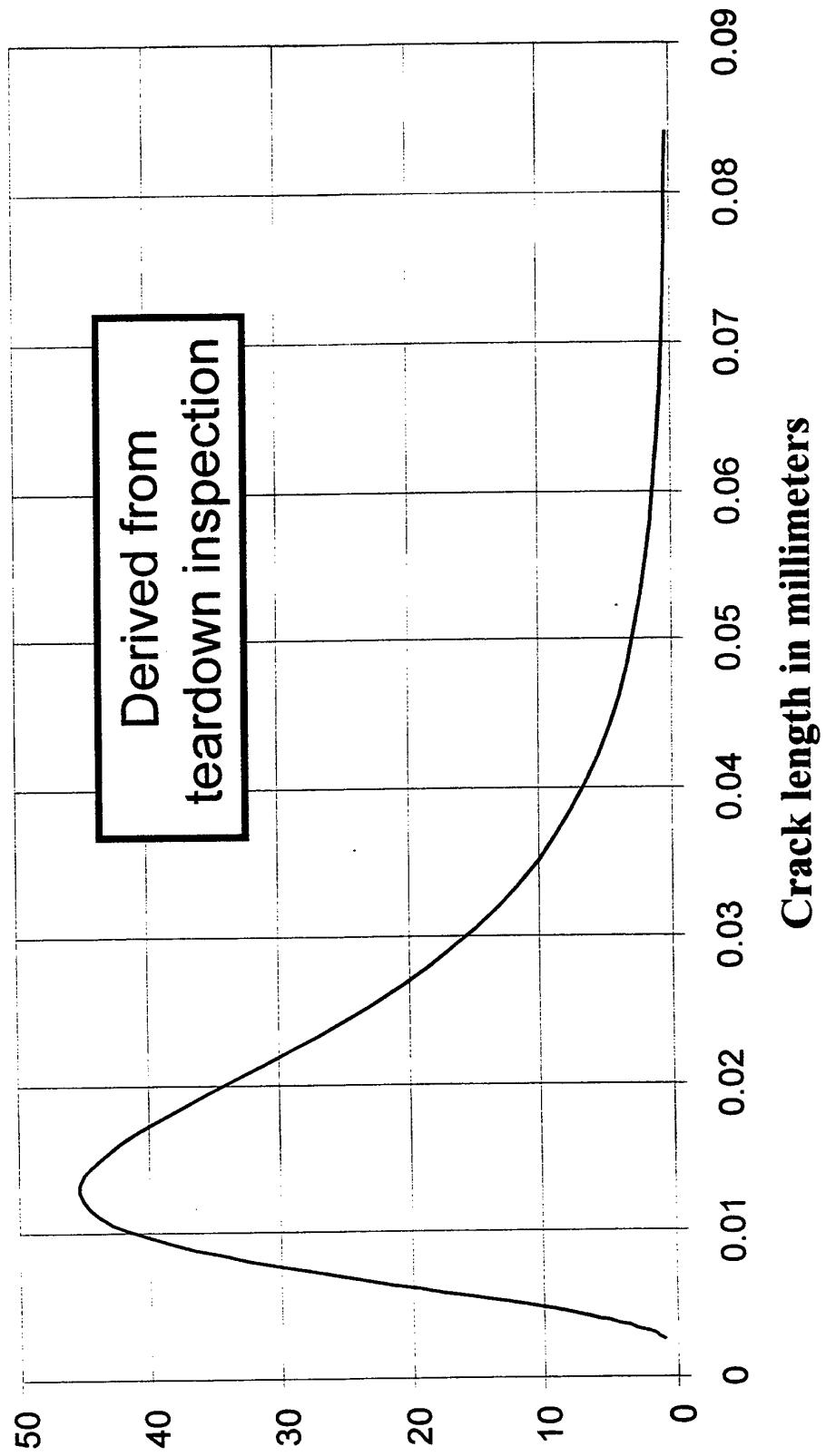
```

## **MAJOR FINDING**

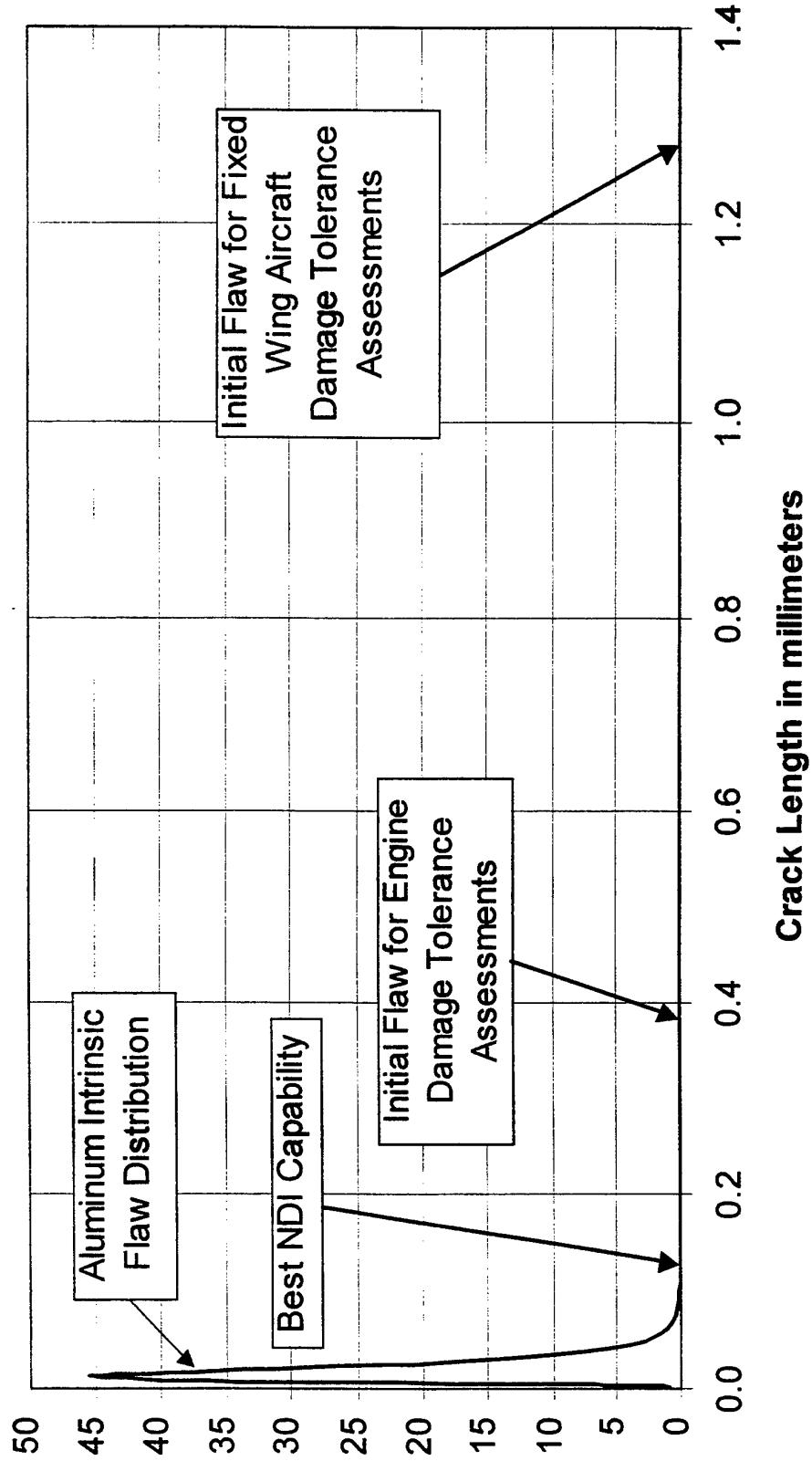
- Original design of the HH-53 based on conventional safe life practice
- Defect distribution for original design representative of intrinsic flaws found in the materials tested

**Nondestructive inspection capability inconsistent with defect distribution used in the initial design**

# INTRINSIC FLAW DISTRIBUTION FOR ALUMINUM



# INITIAL STRUCTURAL FLAWS



## **OTHER FINDINGS FROM THE HH-53 DAMAGE TOLERANCE ASSESSMENT**

---

- Required stress analysis compatible with current technology
- Available stress spectra conservative
- Many critical locations found
  - Although high cycle stresses caused many areas to be not inspectable many areas found to be compatible with inspection capability
- Confidence low in analyses involving crack growth threshold

## OBSERVATIONS

---

- Many parts in existing helicopters are not manageable by damage tolerance methods
- Existing helicopters should be assessed by damage tolerance approach to determine flaw sensitivity of components
- Engine damage tolerance initial flaw sizes are suitable for helicopter design

## OBSERVATIONS (CON'T)

---

- Inclusion of short crack effect is critical for determination of crack growth
  - technology now available to do this
- Effect of stress range truncation should be integral part of damage tolerance assessment
- Shot peening influential in extending lives of component
  - inspection program may still not be viable

## OBSERVATIONS (CONT)

---

### Flaw tolerant design

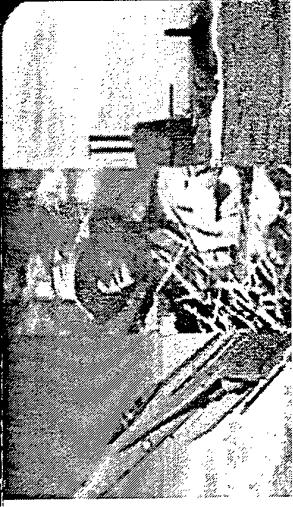
- database insufficient to determine  
determine design flaw size
- difficult for certification authorities to  
determine if the flaws are representative of  
flaws expected in service operations
- as with conventional safe life design the  
flaw tolerant design is based on linear  
cumulative damage

## OBSERVATIONS (CON'T)

---

### Damage tolerant design

- for some components requires reduction in stress that would be adequate for conventional safe life design
- provides for a more balanced design and consequently weight penalty should be small
- fracture mechanics technology available for analyses and tests



# Crack Growth in a Tail Rotor Output Shaft

**Simon Maan**  
Aerostructures Technologies Pty Ltd

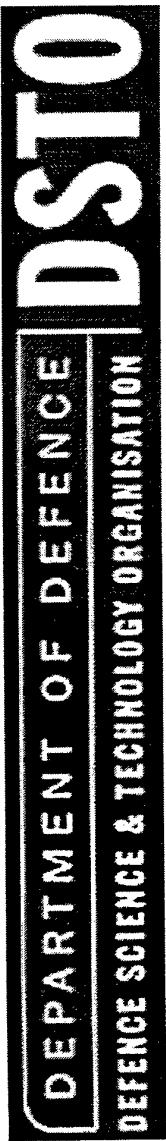
**Domenico Lombardo**  
DSTO

Workshop on Damage Tolerance in Helicopters  
Cranfield, U.K., 4 - 5 April 2000

## OUTLINE

---

- Introduction
- Static Load Analysis
- Crack Growth Analysis
- Conclusions
- Future Work



# INTRODUCTION



## History

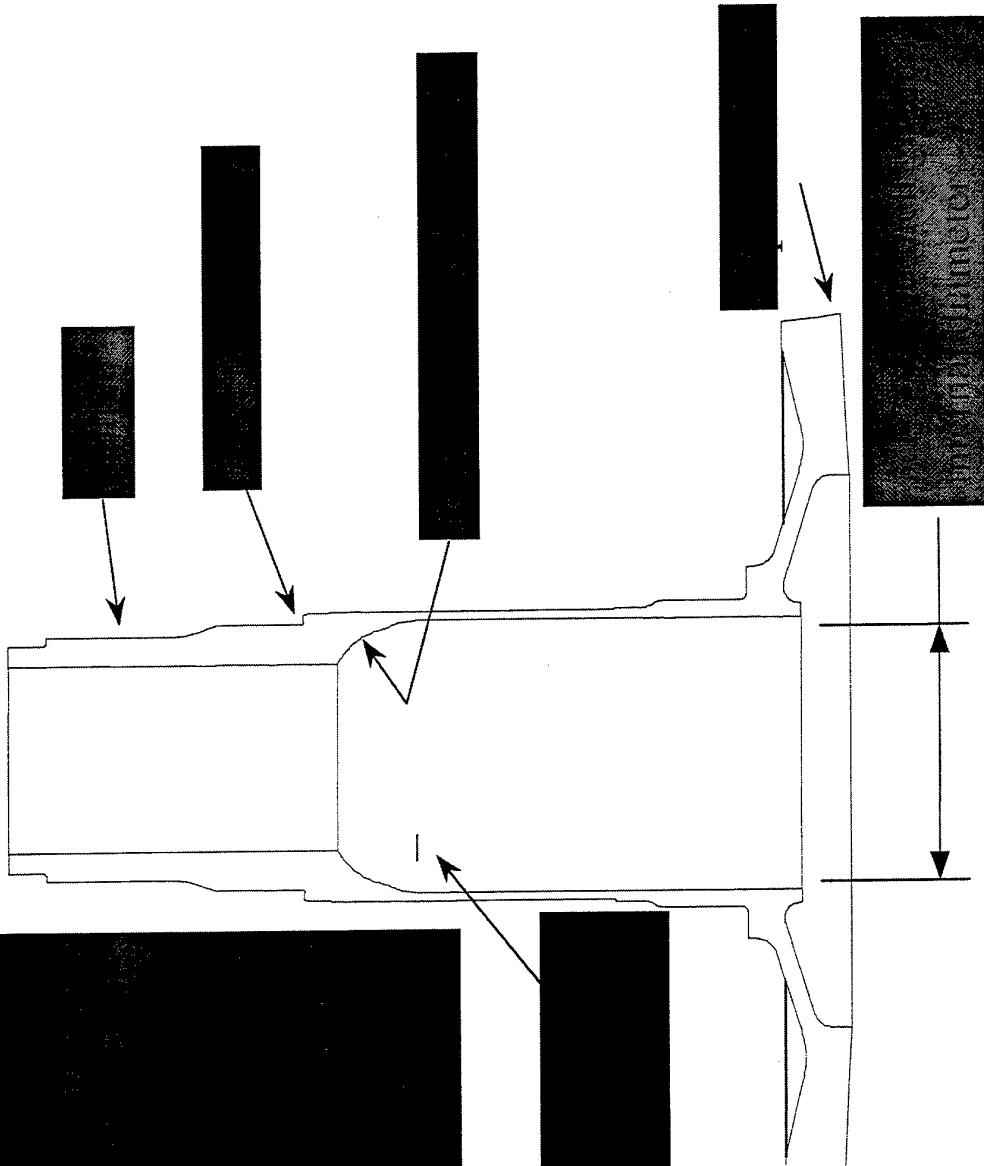
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- On June 9, 1993 a US Army EH-60 Black Hawk crashed due to failure of its TR output shaft
- Prior to this, a Chinese S-70C Black Hawk crashed for the same reason
- The Australian Defence Force, supported by DSTO, took action in 1993/94 to ensure the integrity of the shafts in the Australian Black Hawk and Seahawk fleet

## Failure Details

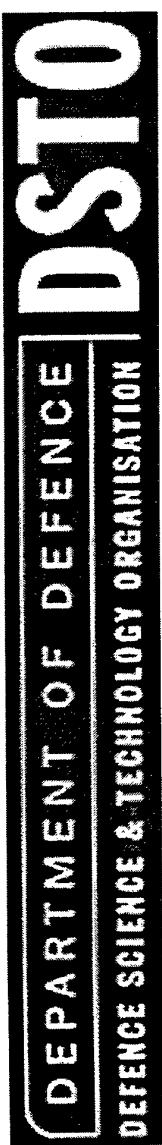
- Both failures due to a crack initiating at an internal radius
- For EH-60, crack initiation was thought to be due to incorrect machining
  - EH-60 shaft: 580 flight hours at failure (5400 hour retirement life)
  - S-70C shaft: 350 flight hours at failure
    - This raised questions about the shaft's tolerance to manufacturing flaws

## Shaft geometry



## Shaft Details

- **Two versions of shaft existed: 70358-06620-101 (“Dash 101”) and 70358-06620-102 (“Dash 102”)**
  - “Dash 101” not shot-peened
  - “Dash 102” Shot-peened
- **Both failed shafts were “Dash 101”**



# STATIC LOAD ANALYSIS

## Objective

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- **Initial Objective:**
  - Determine crack growth rates in the shaft
- **Longer-term objective**
  - Determine time between earliest possible warning if health monitoring had been present and final failure of shaft
  - Determine if health monitoring of the shaft is feasible

## Aerostructures Role

- Aerostructures Technologies Pty Ltd was tasked with:
  - Creating a Finite Element Model (FEM) of the tail rotor output shaft,
  - Performing a crack growth investigation
  - Validating FEM results by experiment

## Modelling of the Shaft

---

- Modelling of the Shaft
  - Four 3D NASTRAN models created using brick elements (Hex 20)
    - > Three models based on geometry of three US Army shafts currently with DSTO
    - > One model based on Sikorsky detail drawings
  - Loads and boundary conditions based on Sikorsky fatigue test of shaft
  - Model results correlated against fatigue test results

## Results of the Investigation

- Stress concentration factors for all four FEMs varied between 1.42 and 1.52
- Peak stresses < 40% of Yield
- Stresses around one inch radius are sensitive to actual geometry of the radius
- Conclusion:
  - Shaft failure in only a few hundred hours difficult to explain if shaft manufactured as per design and no pre-existing flaw is present



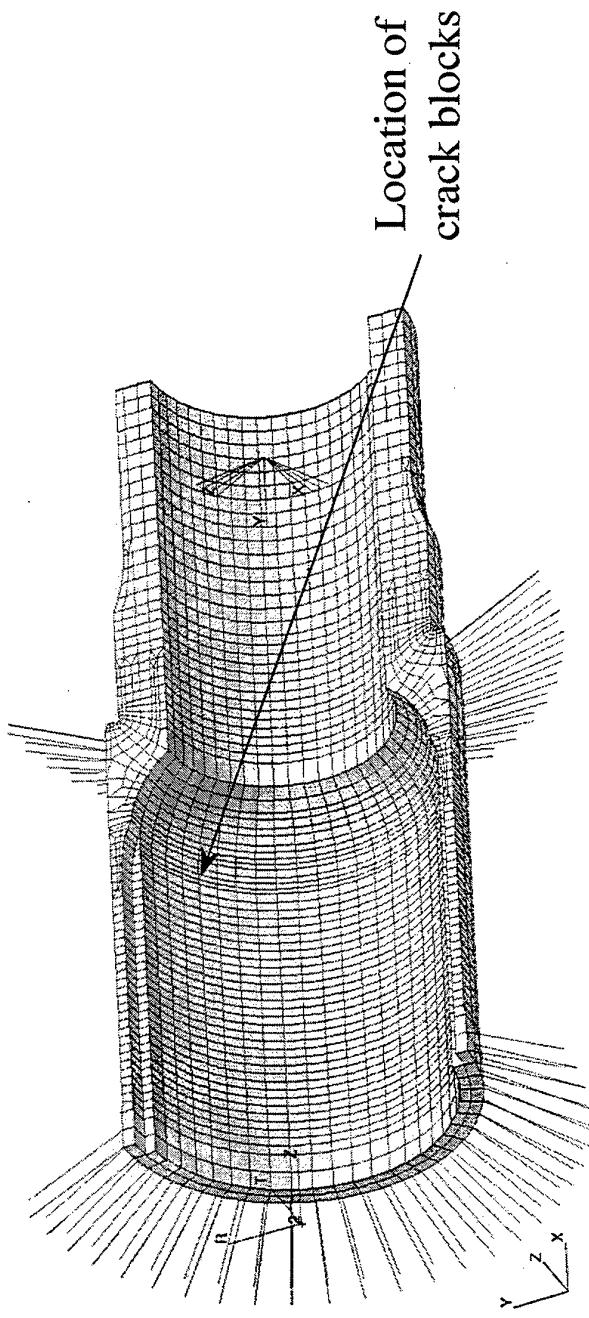
# CRACK GROWTH ANALYSIS

## Crack Growth Analysis

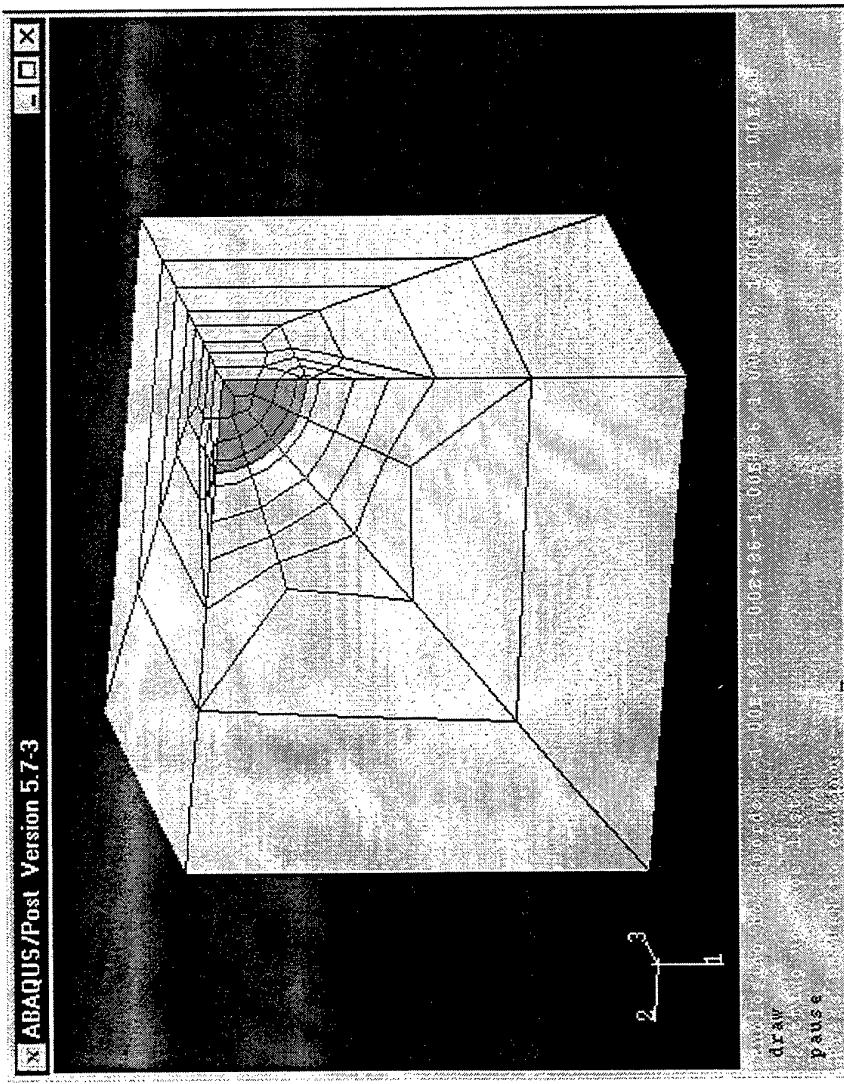
- NASTRAN FEM converted to ABAQUS FE Solver format for use with ZENCRACK
- ZENCRACK allows a 3D crack to be placed into a 3D FE model.
- ZENCRACK also allows the crack to grow in 3 dimensions based upon max stress orientation
- Modified US Army UH-60A/L design usage spectrum used

## ZENCRACK Model

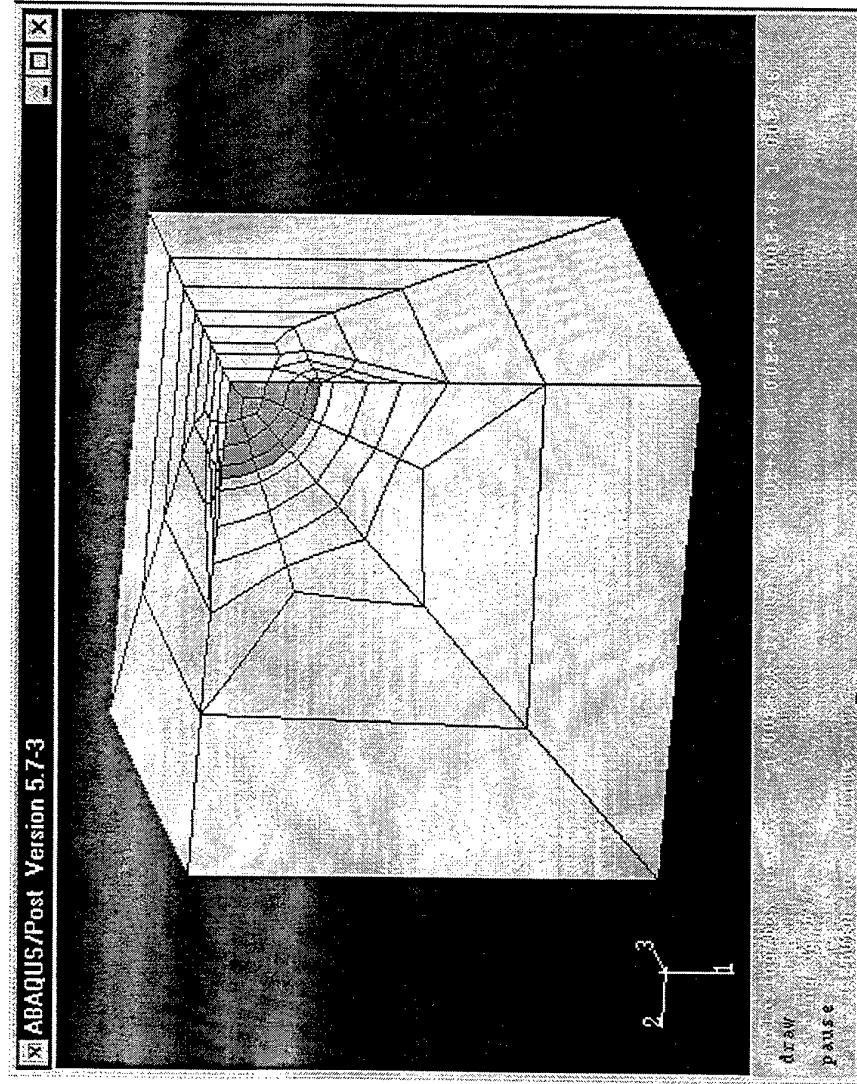
- At the location shown eight crack blocks were inserted into the mesh to model the defect's initial size



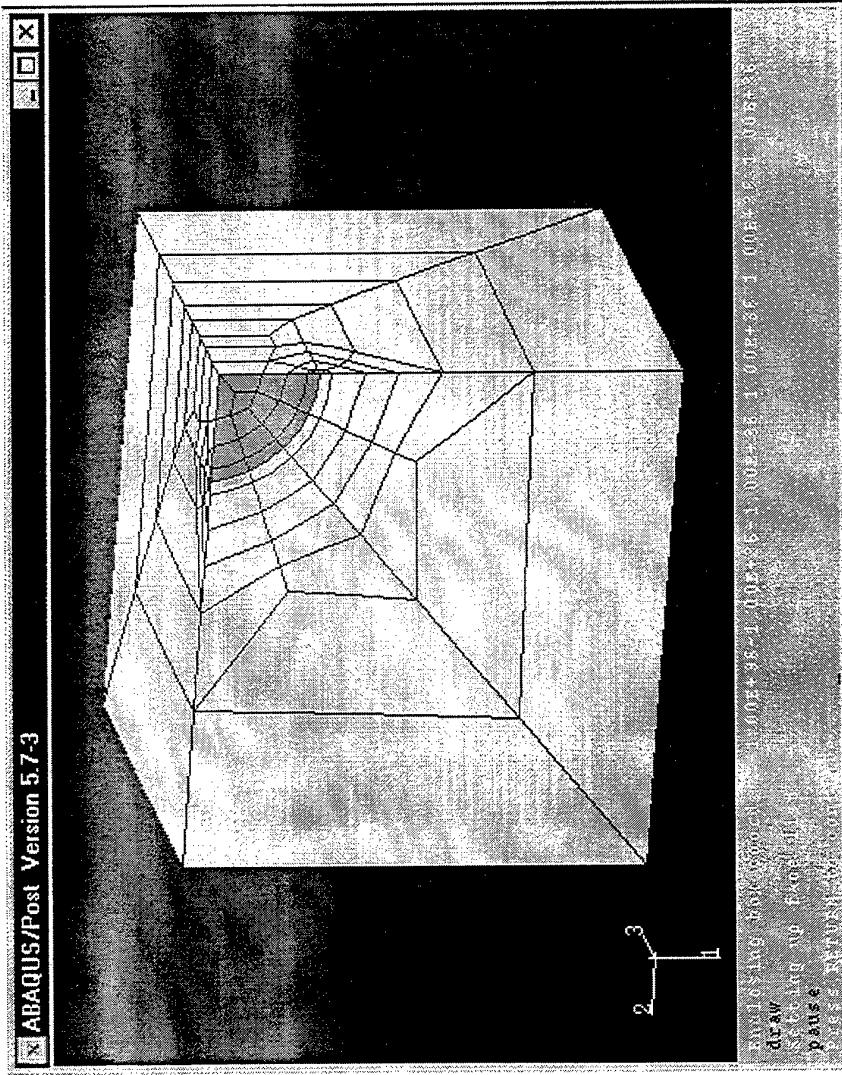
# Crack growth



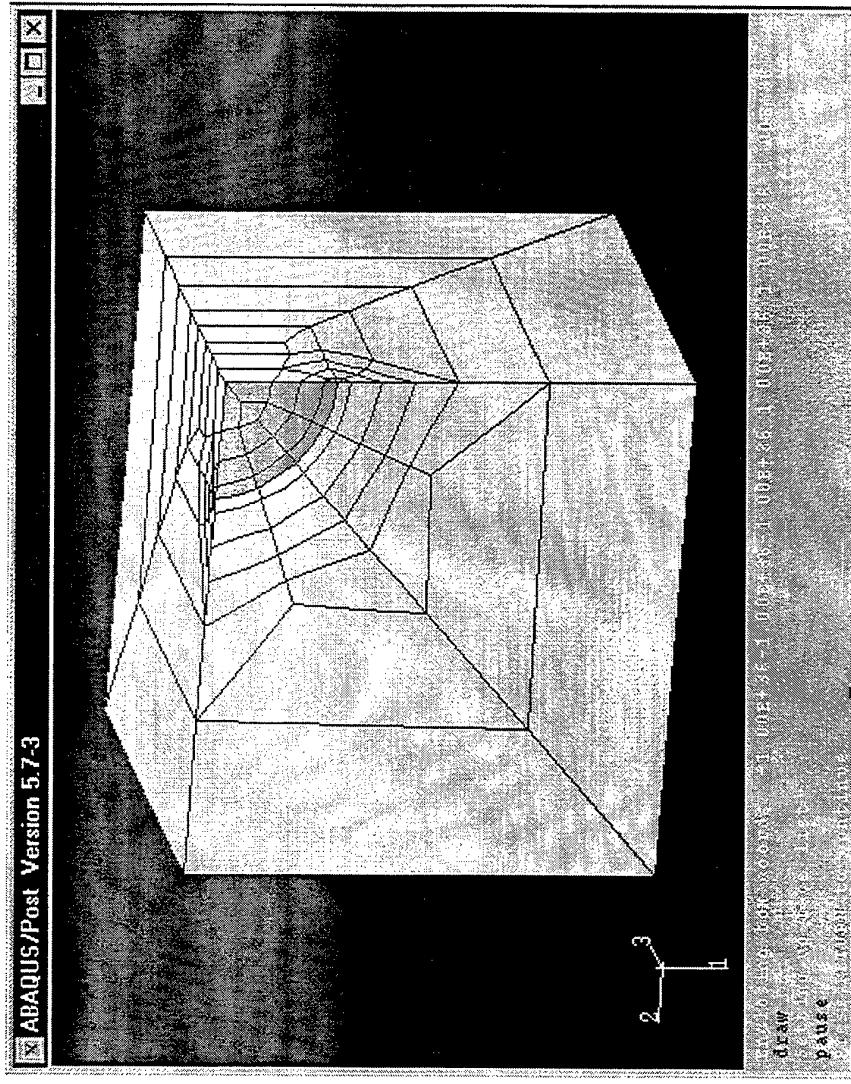
## Crack growth



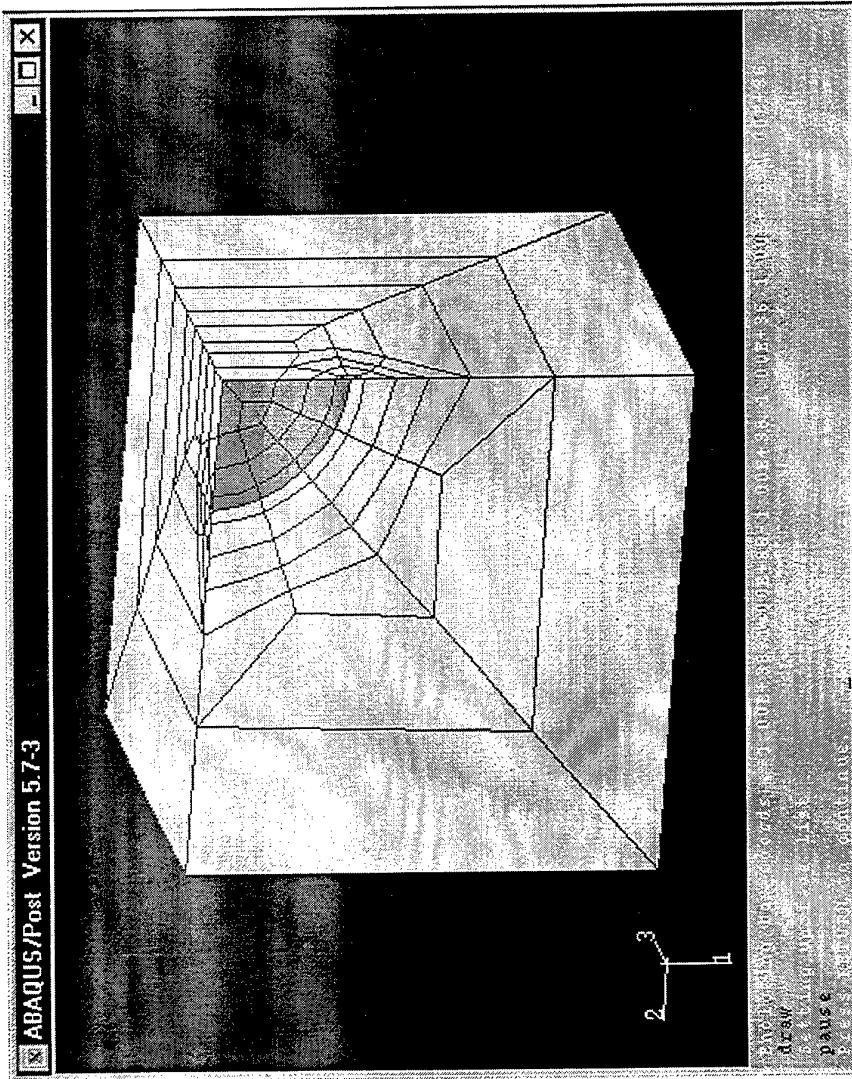
## Crack growth



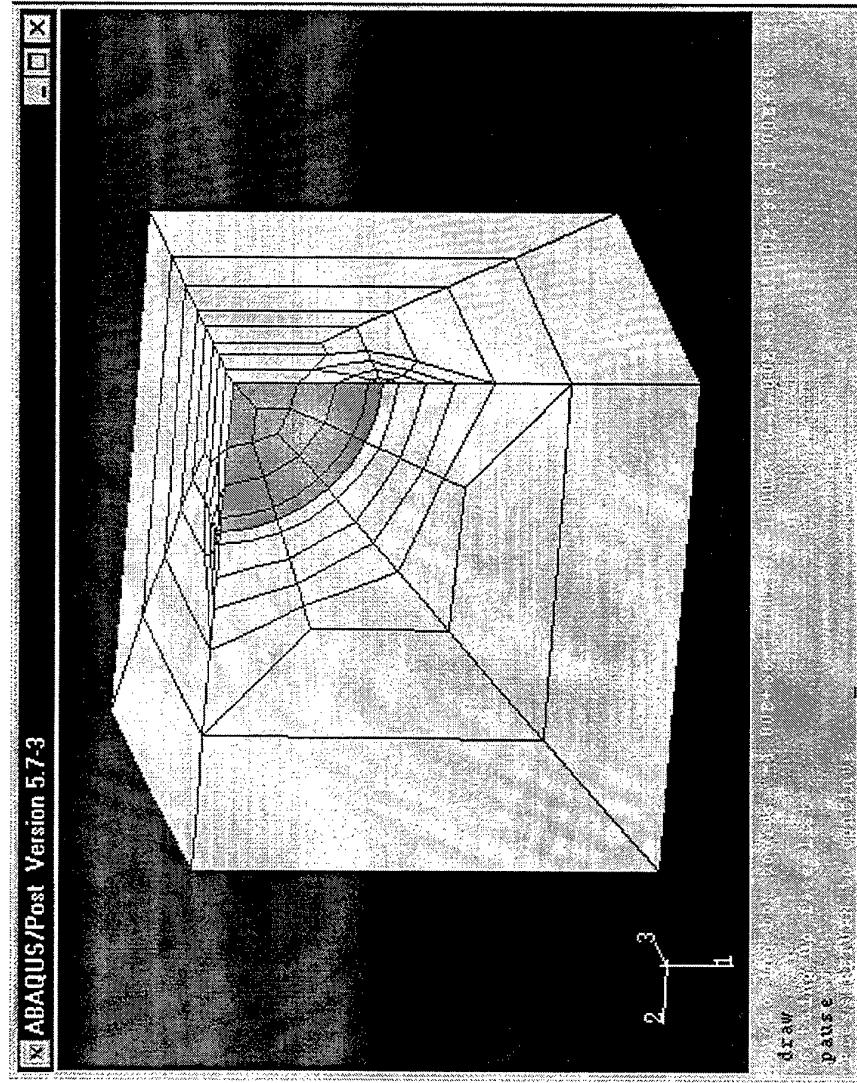
## Crack growth



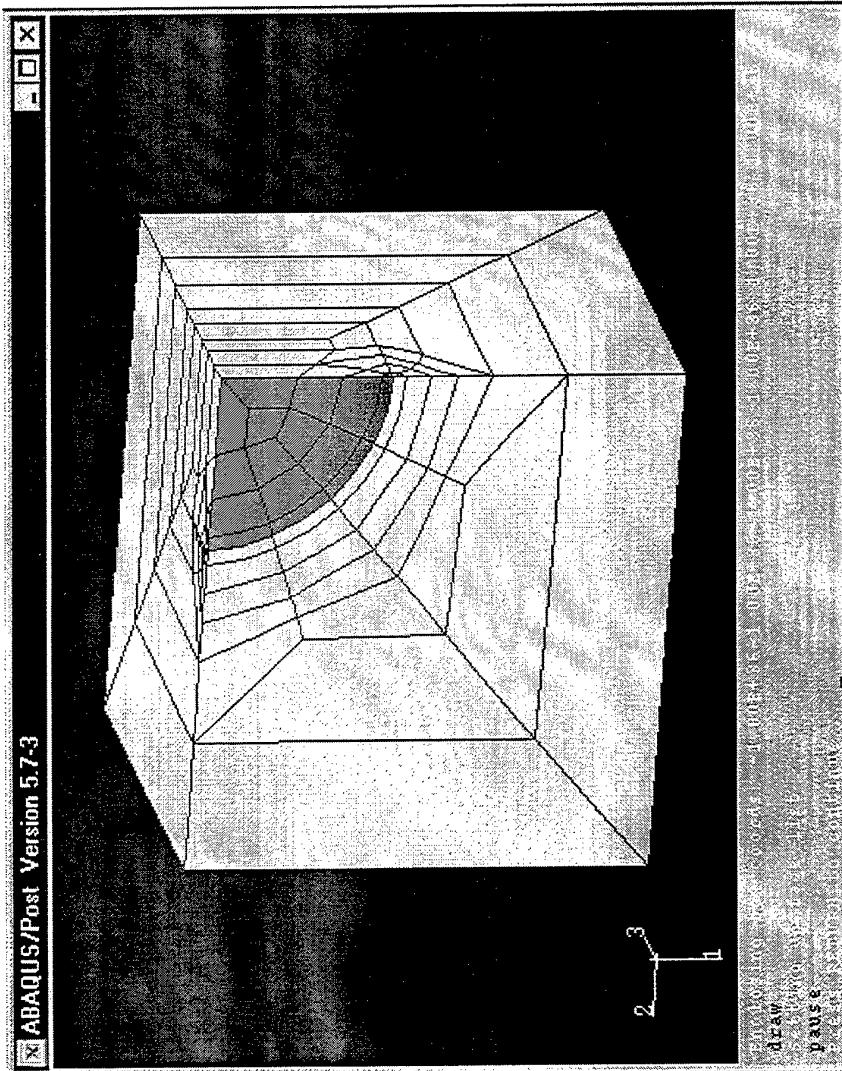
## Crack growth



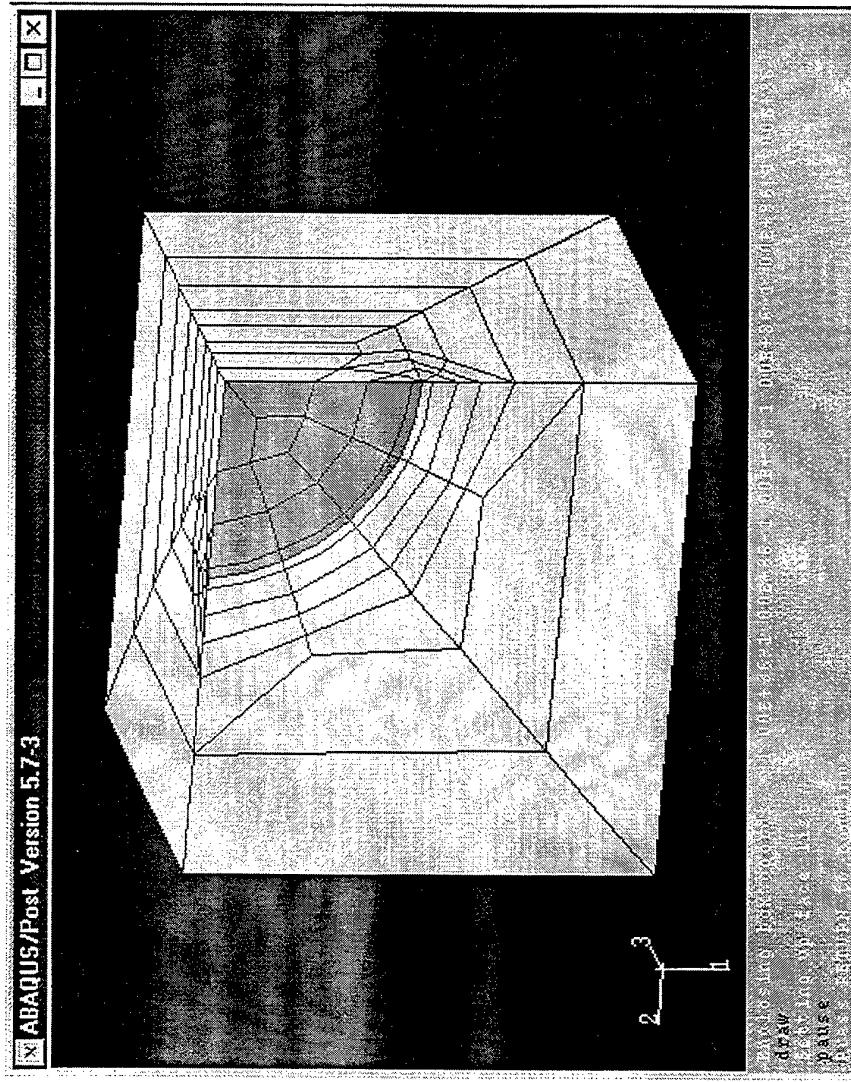
## Crack growth



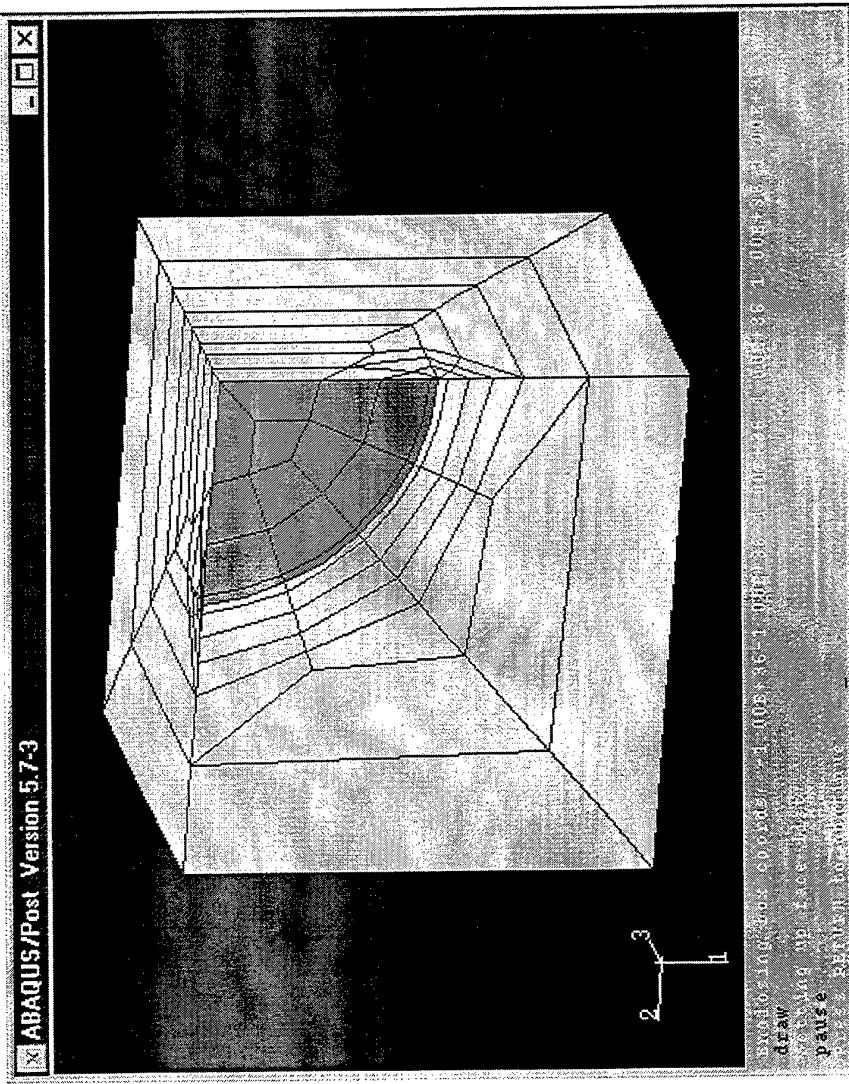
## Crack growth



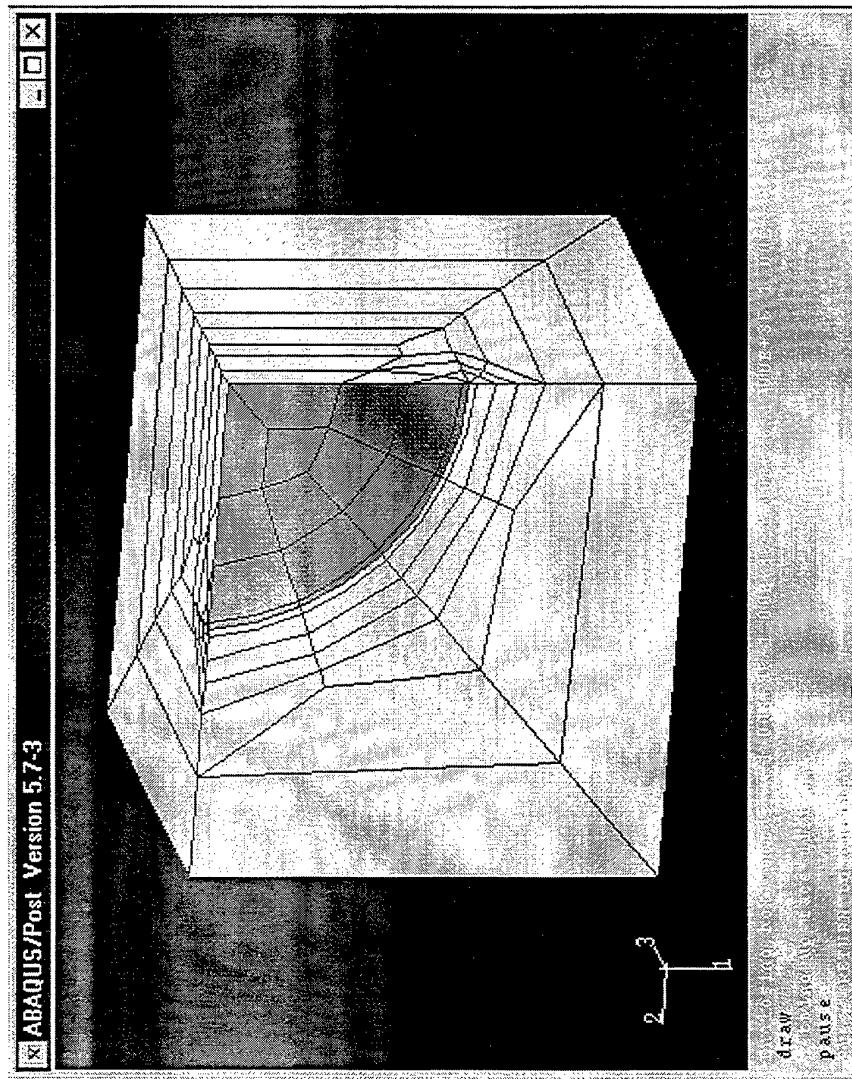
## Crack growth



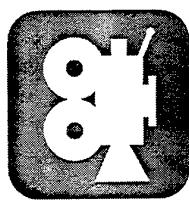
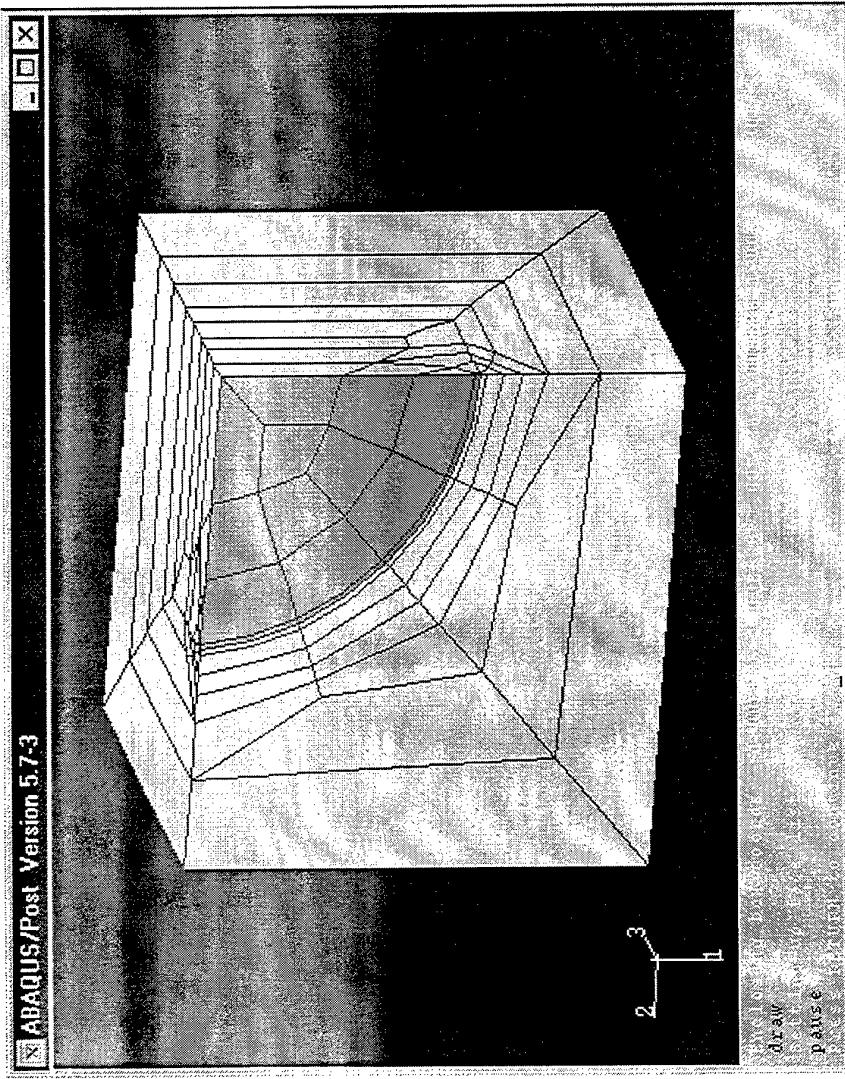
## Crack growth



# Crack growth



# Crack growth

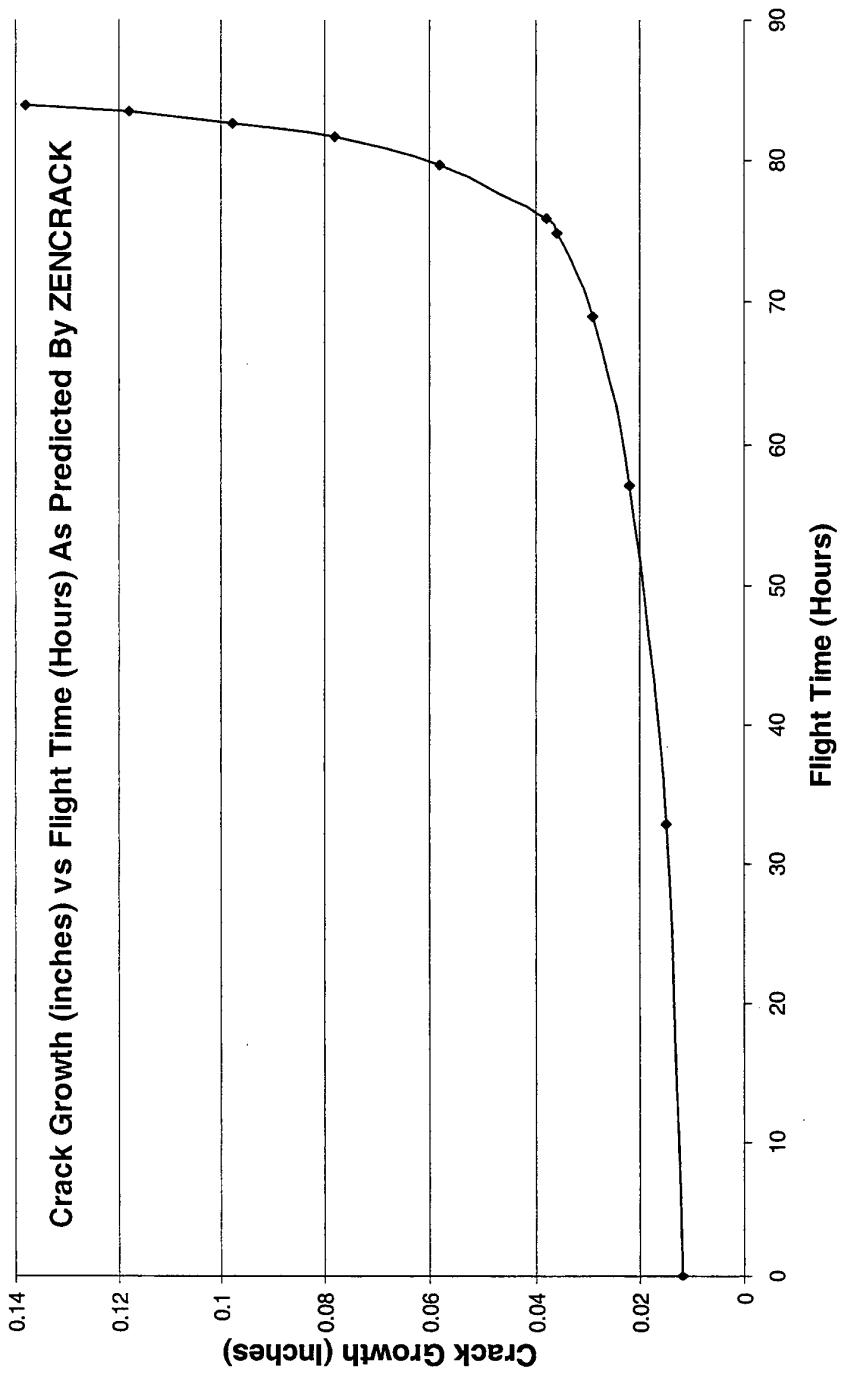


Replay



## ZENCRACK Crack Growth Curve

- The smallest crack that could be modelled measured  $0.012 \times 0.395$  inches (depth x width)



## ZENCRACK Crack Growth Results

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- Time to failure predicted to be  $\approx$  100 to 150 hours
- Compare with 580 hours (EH-60) and 350 hours (S-70C)
- Hence, presence of flaw severely reduces life
- Failure attributed to flaw due to either machining procedure or intergranular corrosion

## ZENCRACK Limitations

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- Problems with applying the usage spectrum
- Only a part of the crack growth history could be modelled
- Therefore couldn't completely quantify results

## AFGROW Crack Growth Analysis

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- AFGROW was used to provide a comparison with the ZENCRACK results
  - Semi-circular element used
  - Applied max loads from the four NASTRAN FEMs to the element boundary
  - Applied an unclipped load spectrum
  - 3D model, but 2D crack growth

## **AFGROW Crack Growth Results**

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- Similar results to Zencrack
- 67 hours to grow the crack from 0.03 inches x 0.19 inches to a crack that penetrated the outer wall of the shaft (0.233 inches)
- AFGROW would not initiate a crack  $< 0.03$  inches

## FEM Experimental Validation

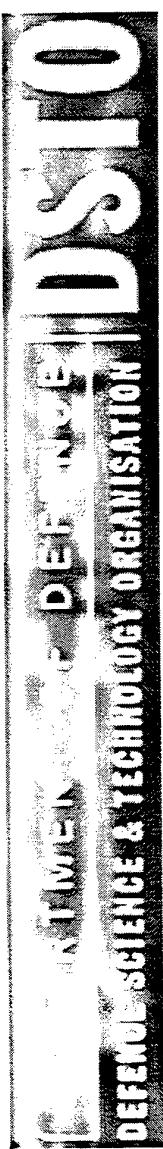
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- Correlation so far based on Sikorsky fatigue test results
- However:
  - Insufficient strain data exist for proper correlation
  - Correlation performed at locations of applied load (possible local stress effects)

## Static Strain Survey

- NASTRAN FEM results will be verified by experimental testing
- Static strain survey of the shaft to be done
- Shaft has strain gauges at locations near to the crack initiation site
- Results pending

# CONCLUSIONS

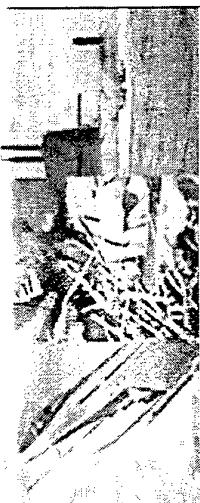


## Conclusions

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- Tail rotor output shaft is sensitive to flaws near the internal one inch radius
- Large life reduction results from presence of flaw
- Shot-peening beneficial

INDIA'S SCIENCE & TECHNOLOGY ORGANISATION | DST



## FUTURE WORK

## Future Work

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- Ideally, perform crack growth testing on shafts to validate ZenCrack and AFGROW results
- Determine when a health monitoring system might have given warning of impending failure
- Determine feasibility of health monitoring of shaft

# SUMMARY

# DIMENSIONS



## Summary

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- TR Output Shaft failures in Black Hawks
  - Failures at low number of flight hours (<< retirement life)
- Static FEM analysis
  - Showed stresses insufficient to cause such early failures
- Crack growth analysis
  - Flaws in shaft internal radius significantly reduce life

**Thank you for your time**

**PRESENTATION TO**  
**WORKSHOP on DAMAGE TOLERANCE in HELICOPTERS**

**4 - 5 APRIL 2000**

**APPLICATION OF DT TO HELICOPTER  
CERTIFICATION**

J. Roesch  
MTU Helicopters Inc

1

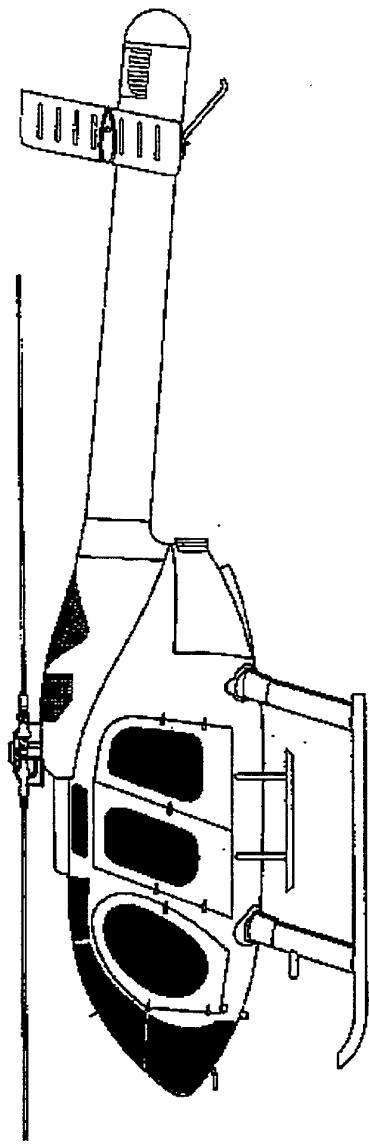
Workshop on DT in Helicopters  
Cranfield University, UK 4/4-5/4/2000

## OVERVIEW

- 600N CERTIFICATION
- FATIGUE & DT METHODOLOGY
- CONTROL SYSTEM COMPONENTS
- INITIAL FATIGUE EVALUATION
- DAMAGE TOLERANCE EVALUATION
- FATIGUE TESTING
- CONCLUSIONS

**COMPOSITE  
COMPONENTS**

Fan Strap  
Vertical Stab  
Thruster



**METAL-COMPOSITE**

Fan Blade  
Tailboom  
Horizontal Stab

**MD 600N**

**CERTIFICATION BASIS**

| <b>Cert Basis</b> | <b>Fatigue (Safe-Life)</b> | <b>Damage Tolerance</b>                     |
|-------------------|----------------------------|---|
| 600N              | FAR 27                     | All<br>Composites /<br>M/R Blade / Controls |

# BASIC METHODOLOGY -- LOADS

(Fatigue and Damage Tolerance)

- AC No. 20-95 (Modified for specific aircraft)
- Flight Strain Survey of all specific maneuvers
  - (3 Altitudes and 4 Gross weights -- Flown aggressively)
- Cycle Count all data
- Multi-mission spectra
  - Worst case Training and Corporate
  - External Load Use (Logging)
- Use MS to define Stress Spectrum for Control System Components

## BASIC FATIGUE METHODOLOGY -- METALS

- AC No. 20-95
- Reduction Factor taken on S/N curve strength  
CAM 6 - Appendix A
- Reduction Factor taken on Mean Test Derived S/N curve  
AFS 120-73-2  
AFML-TR-69-65, "Reliability Analysis Approach to Fatigue  
Life Variability of Aircraft Structures"
- Fatigue tests either spectrum or constant amplitude (spectrum  
preferred)
- Minor's cumulative damage

## DAMAGE TOLERANCE CERTIFICATION METHODS FOR METALS

- NASA/FLAGRO\* program and database used for all calculations
- Crack growth substantiated inspection intervals for M/R Blade
- Crack growth analysis used on control system components for initial certification

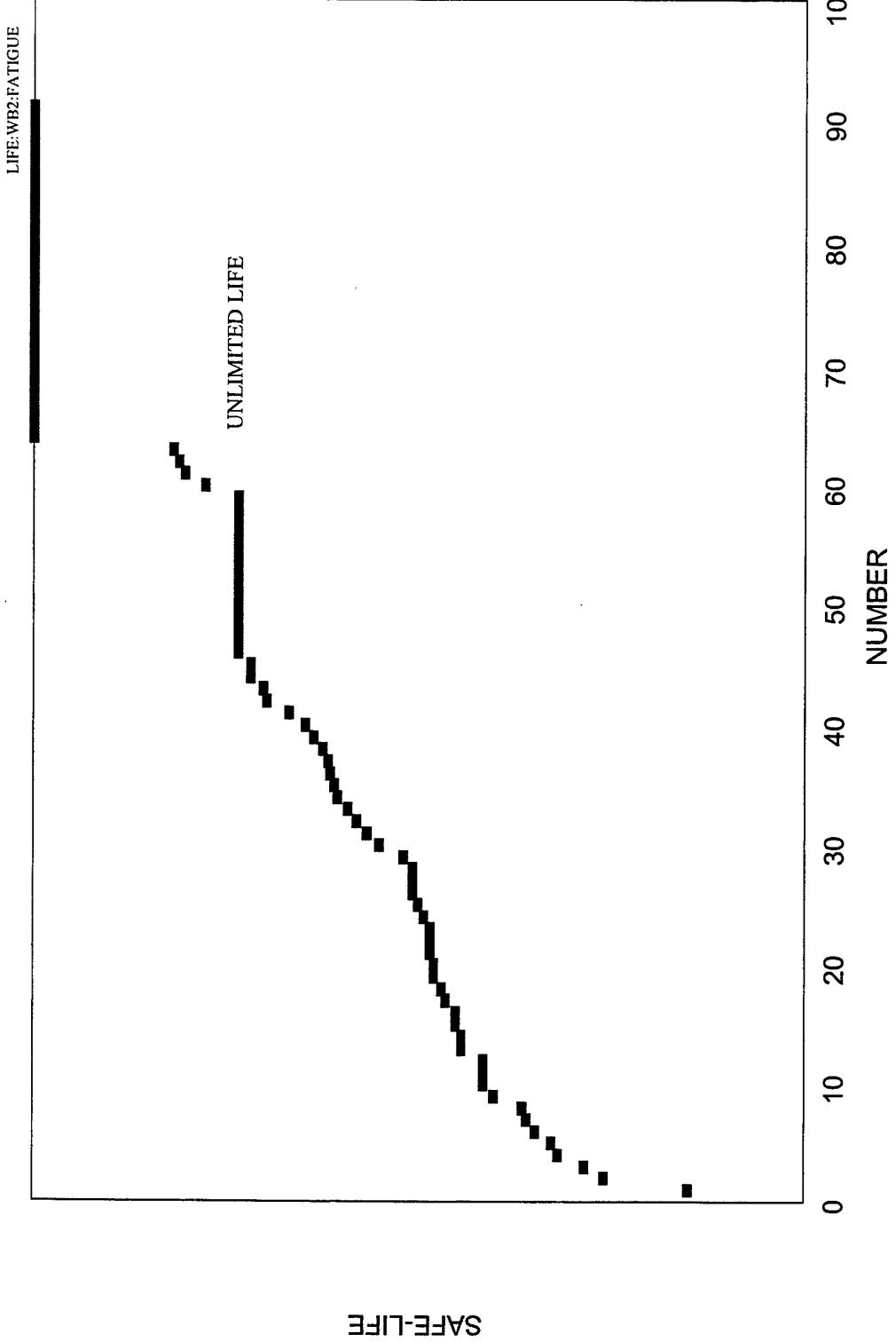
\* NASA Lyndon B. Johnson Space Center Houston, Texas

## INITIAL FATIGUE EVALUATION

- Ground Rules Were To Use Existing Components
- Very Few Fatigue Tests Conducted On Existing Components
- Conservative Factors Required for Analysis Only Certification

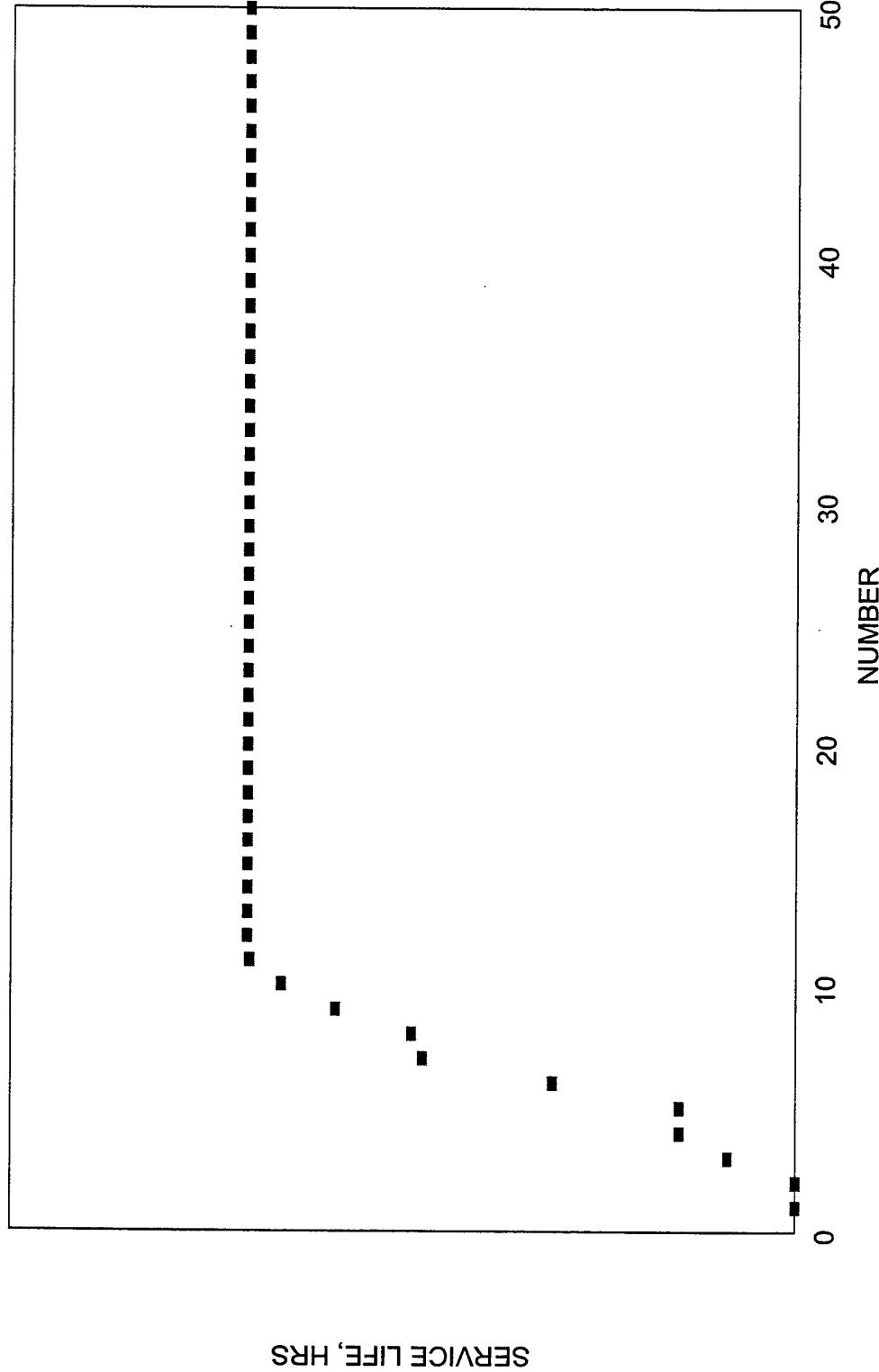
## INCREASED CONTROL SYSTEM LOADS

- Six Main Rotor Blades Instead of Five
- Revised Swashplate & Pitch Housing Geometry
- Increased Weight
- Increased Power

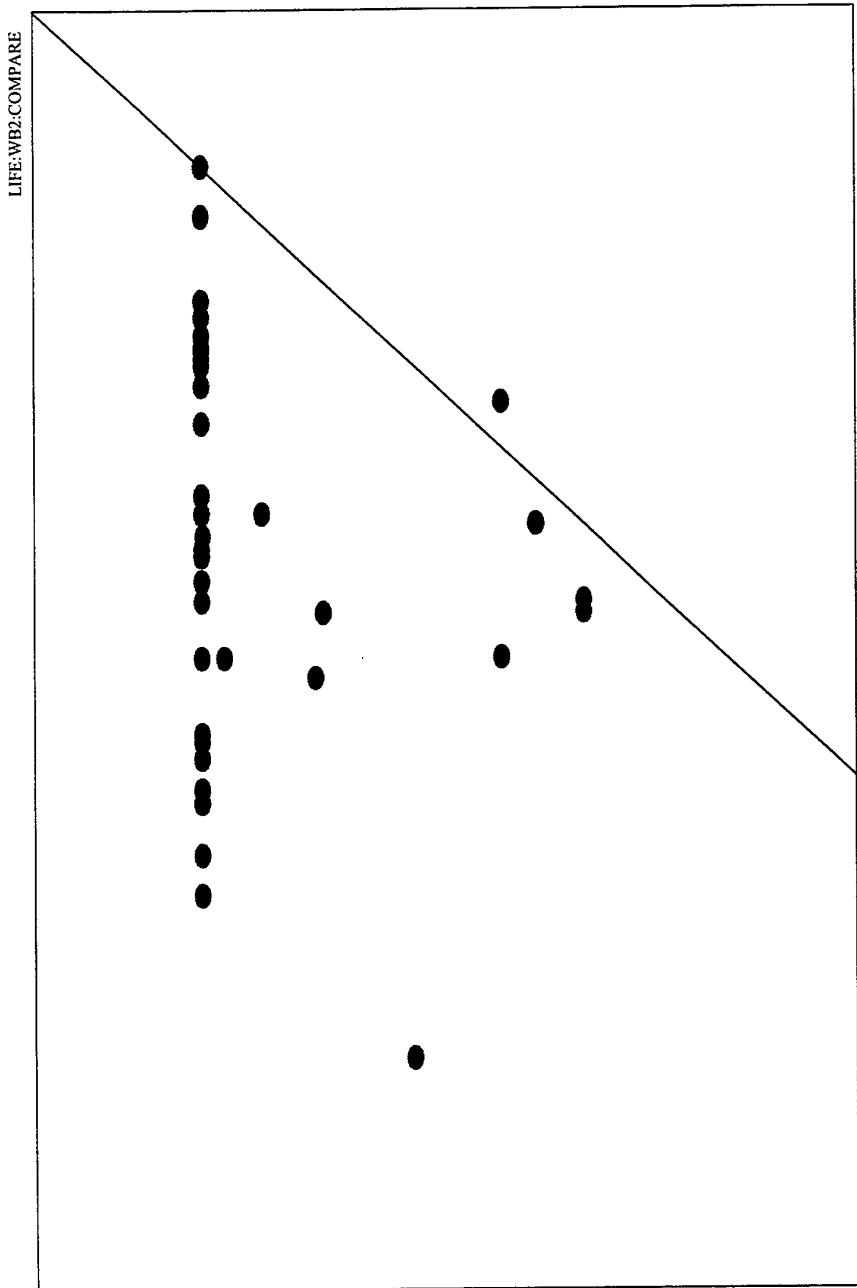


## DAMAGE TOLERANT EVALUATION

- Select Equivalent Initial Flaw and Size  
( $a = .02$  to  $.05$  inch Deep Corner or Surface Flaw)
- Use Same Fatigue Spectrum
- Account for Load Transfer with Bearing Stress
- Apply Factor of Four (4.0) to Calculated Life



SERVICE LIFE FROM DT



## DAMAGE TOLERANT RESULTS

- Reduced Number of Life Limited Components from 43 to 9
- Service Life Ratio (DT / SL) ranged from 700 to 0.67
- Identified 8 Components for Redesign
- Reduced the Number of Fatigue Tests to 6

## FATIGUE TEST RESULTS

- Fatigue Testing Showed 7 of the 9 Components to have Unlimited Life Using Safe-Life Methodology
- Fatigue Testing of the Lowest Life Component Confirmed Low Life (Service Life Recalculated Using Safe-Life Methods)
- Other Low Life Component was not Tested - Redesigned

## **CONCLUSIONS**

- Damage Tolerance Successfully Established Service Lives on 600N
- For "Analysis Only", DT Lives are More Accurate than Safe-Life Lives
- DT Lives are Conservative Compared to Final Safe-Lives in Majority of Cases



## **Research Issues in the Application of Damage Tolerance Concepts to Rotorcraft**

Presented by

**Mel F. Kanninen  
Galaxy Scientific Corp.**

Based on Work in Conjunction with

**Charles Harrison, FAA Rotorcraft Directorate  
Dy Le, FAA Technical Center  
Sharon Miles, FAA Rotorcraft Directorate**

Presented at

**Workshop on Damage Tolerance of Helicopters  
Cranfield University, UK  
4-5 April 2000**

# **Current and Anticipated FAA Regulations for Normal and Transport Category Rotorcraft**

**Acceptable fatigue tolerance evaluation  
procedures (alone or in combination)  
currently in CFR Parts 27.571 and 29.571:**

- Damage Tolerance
- Flaw Tolerant Safe-Life
- Fail-Safe
- Safe-Life

**Based on TOGAA recommendations, goal  
is to allow only damage tolerance and safe  
life determinations starting in year 2003**

# **Basic Elements of the Damage Tolerance Methodology for Addressing Crack Growth Failures**

- Reliable and precise NDI Procedures that are applicable to all PSE's and all possible crack sizes, shapes and orientations
- Reasonable estimations of the expected service Loading Spectra, and a reliable truncation (simplification) procedure
- Material Fatigue Crack Growth Characterization and other material properties associated with the anticipated modes of cracking
- Robust (efficient, flexible & accurate) Fracture Mechanics Analysis procedures for crack growth and residual strength determinations

# Technical Challenges to General Use of DT for New and Existing Rotorcraft

- Complex fuselage and drive train geometries makes the use of conventional finite element models very cumbersome and/or inaccurate
- Rapid accumulation of load cycles requires consideration of very small crack sizes that can be well outside the range of validity of LEFM
- Surface treatments that retard fatigue by creating compressive residual stress fields (e.g., shot peening) require inelastic fracture mechanics
- Mission load profiles that reflect the true severity of actual operating conditions (i.e. combined HCG and LCF load spectra)
- Blunt initial damage (e.g., tool marks, corrosion pits, mechanical impact) for which safe life is nonconservative but LEFM is overconservative
- Composite materials require consideration of failure modes and NDI techniques beyond those of LEFM developed for metallics

## **Summary and Conclusions**

- 1. Fracture mechanics based damage tolerance methodology currently provides the only practical and scientifically sound approach to establishing in-service inspection intervals for rotorcraft.**
- 2. A new rule embodying damage tolerance as a complement to safe life, with the goal of eliminating flaw tolerance safe life, will be implemented by the FAA by the year 2003.**
- 3. Because of the more demanding conditions facing fracture mechanics analyses in rotorcraft, there is a need to generalize the currently available LEFM-based DT methodologies**
- 4. In collaboration with RITA through the NRTC, with technology transfer from transport aircraft and engines, the FAA is performing research to enable DT to be more widely used for rotorcraft**

## **Additional Research Issues that if Addressed would Enhance the Use of DT for Rotorcraft Applications**

- Incubation/formation of small cracks
- crack growth in mixed mode conditions
- crack initiation from initially blunt damage
- crack growth in residual stress fields
- combined corrosion and crack growth
- probabilistic methods to treat uncertainties
- EIFS determinations to link safe life and DT

## **SESSION C**

### **USAGE, HEALTH MONITORING & INSPECTION**



*Cranfield*  
UNIVERSITY



**DERA**



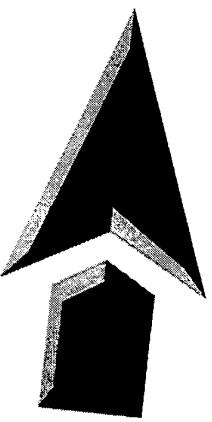
# *Helicopter Fatigue Life Monitoring in the 21st Century*

Workshop on Helicopter Damage Tolerance,  
4-5 April 2000, Cranfield, UK

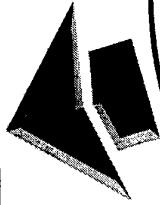
A.A. ten Have  
**National Aerospace Laboratory NLR**  
**Structures and Materials Division**  
**tel: + 31 527 248292**  
**fax: +31 527 248210**  
**e-mail: have@nlr.nl**

# Life Cycle

## OPERATIONAL USAGE



Usage Monitoring  
is Looking at...



## REMEDIES

sched. maintenance  
periodical inspections  
finite lives  
replacements  
modifications  
repairs

## DEGRADATION

time  
loads  
environment  
(corrosion, erosion,  
humidity,  
temp. etc.)

... Safety and  
Economy

# Fatigue Life

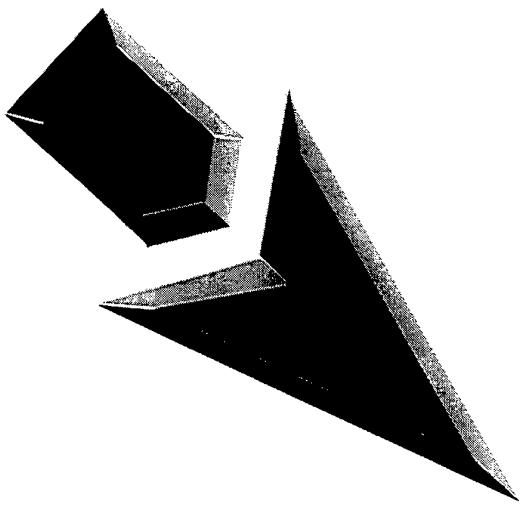
## ● On-Condition Items

- \* 'Damage Tolerant' design
- \* periodical inspections and repairs (if necessary)
- \* non-finite fatigue life
- \* finite economical life
- \* inspection intervals based on operational usage estimate  
(done by manufacturer)

## ● Lifed Items

- \* finite 'Safe Life'
- \* no in-service inspections
- \* based on statistics (99%/95%-rule)
- \* finite life based on operational usage estimate  
(done by manufacturer)

# Fatigue Life



Typical for  
helicopter  
components

- Lifted Items

- \* finite 'Safe Life'
- \* no in-service inspections
- \* based on statistics (99%/95%-rule)
- \* finite life based on operational usage estimate  
(done by manufacturer)

# Fatigue Life calculation

## material testing

- full scale components
- sub components
- coupon specimens

reduction factor for  
variance in testresults

'safe life' in number  
of cycles

**Safe  
Life  
(FH)**

## design loads

- mission type/mix
- events per mission
- loads per event

reduction factor for  
variance in usage

safe 'exchange rate'  
in cycles/FH

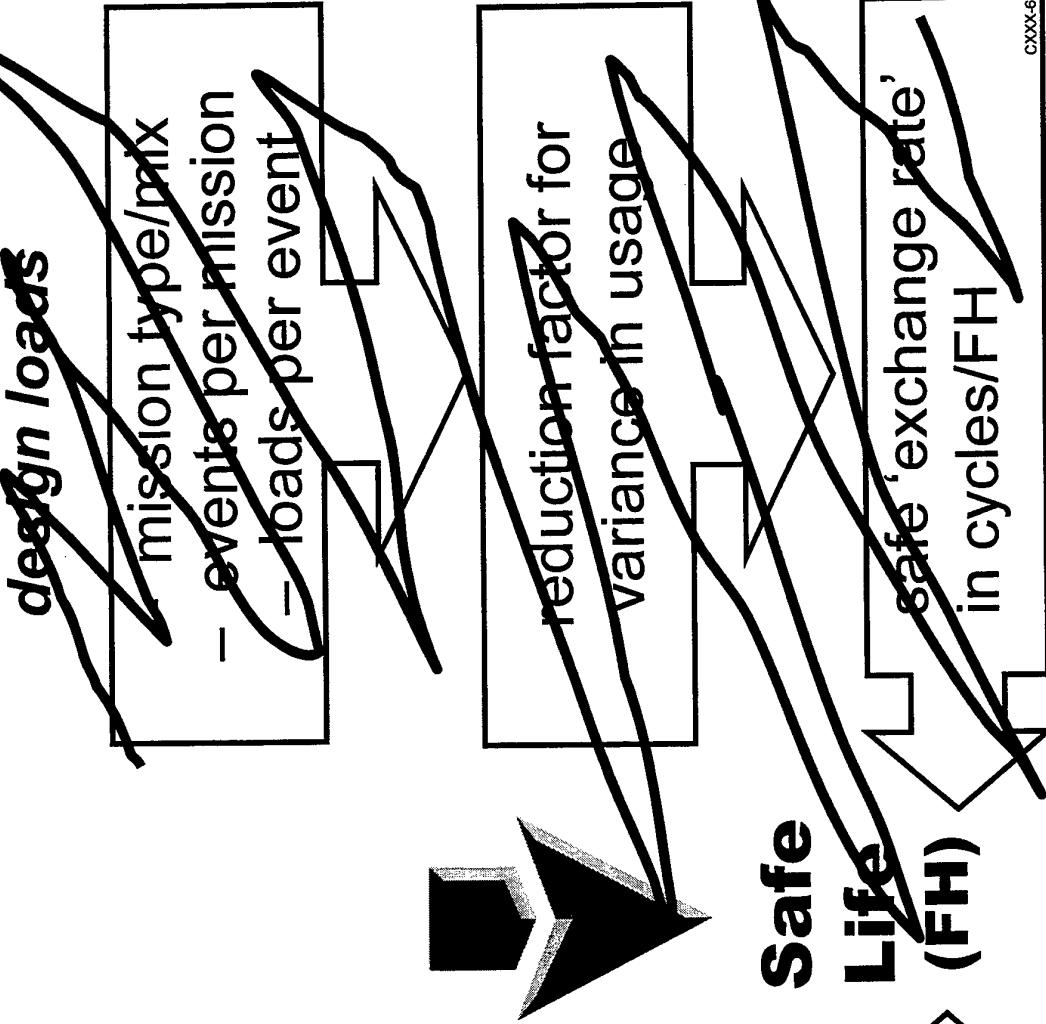
# Fatigue Life calculation

## material testing

- full scale components
- sub components
- coupon specimens

reduction factor for variance in test results

'safe life' in number of cycles



# RNLN Fatigue Life Monitoring...

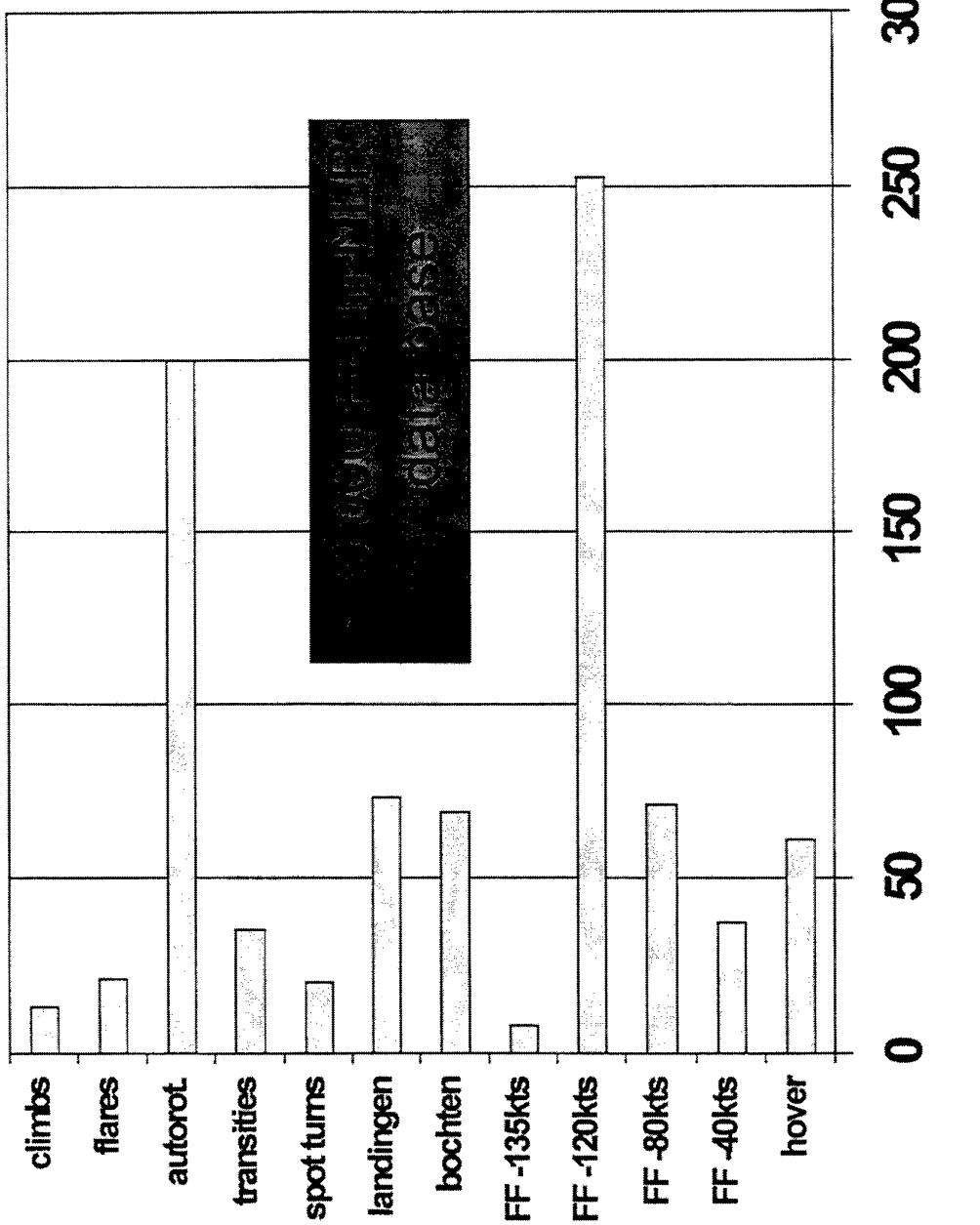
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- Lynx Usage Monitoring

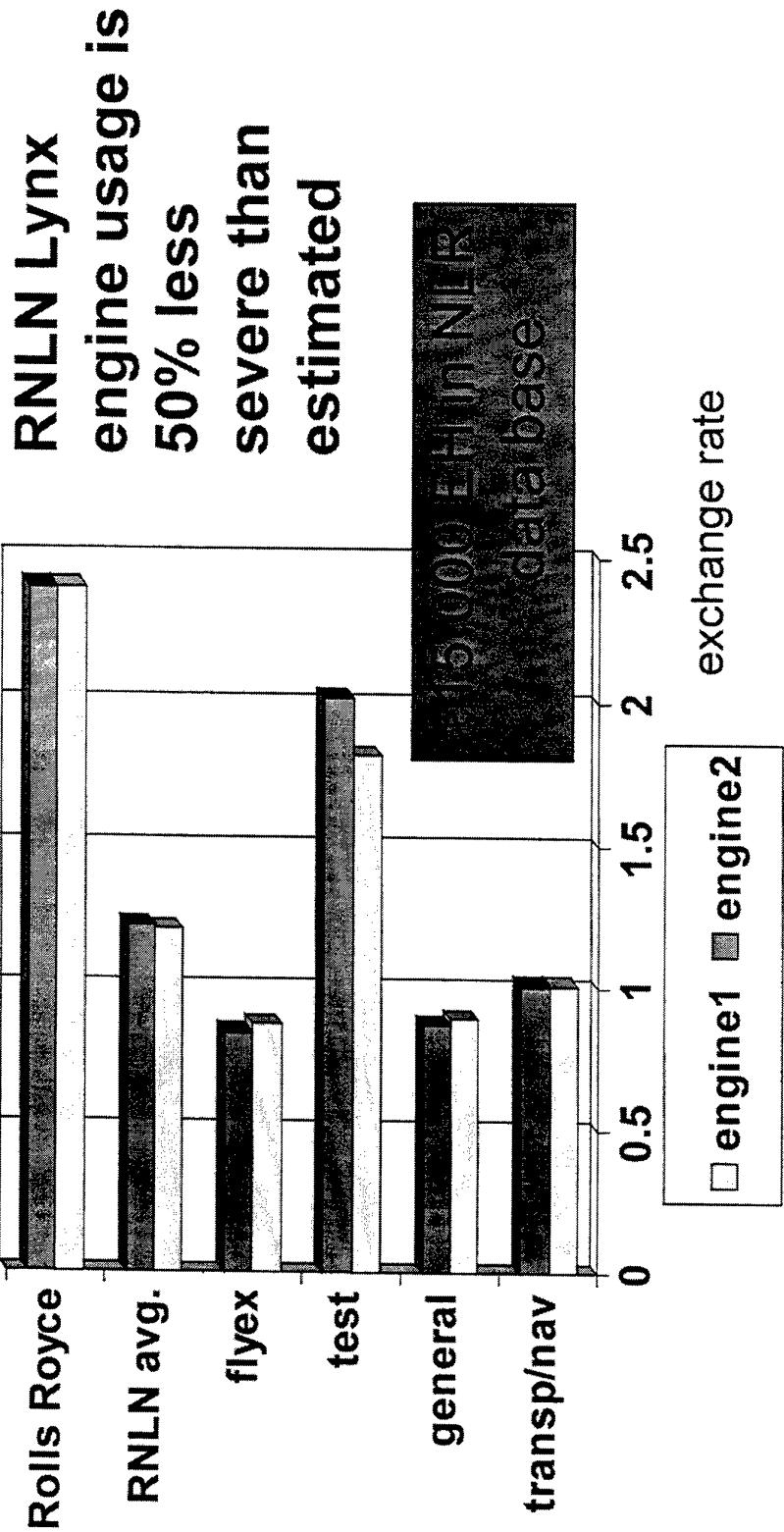
- Lynx Gem Cyclic Life Control
  - \* Rolls Royce Gem engine

- Lynx AIDA
  - \* multichannel retrofit system

# Lynx Usage Monitoring



# Gem Cyclic Life Control



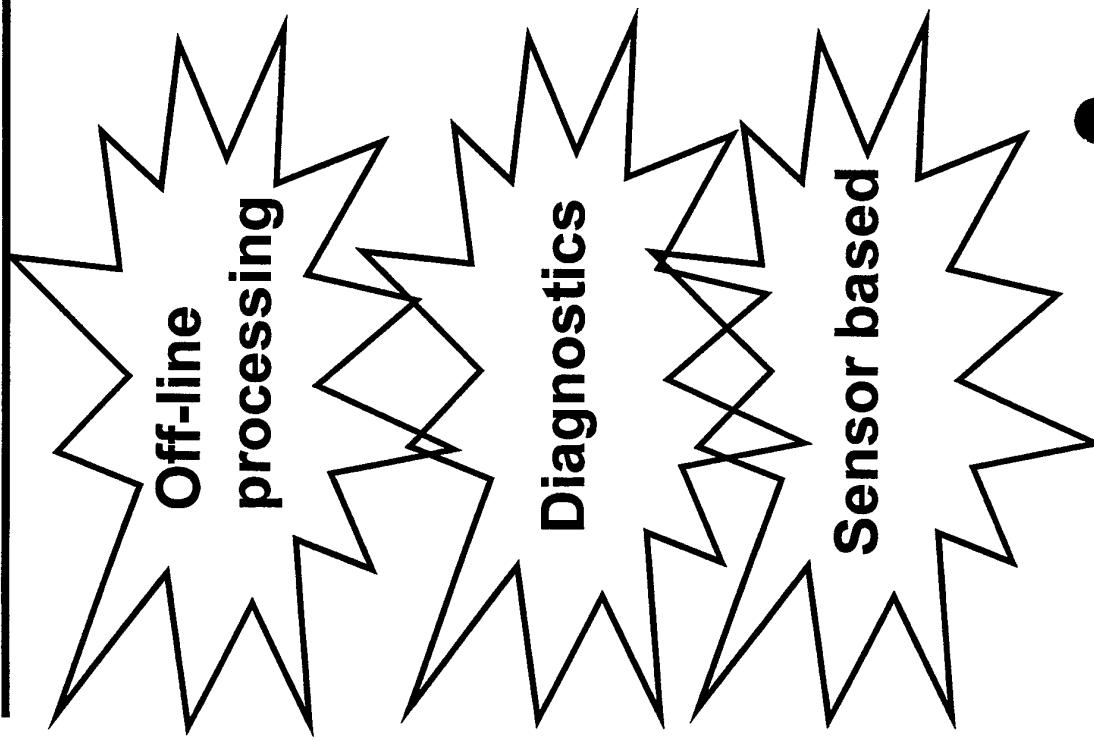
# Lynx AIDA

- **Nr monitoring**
  - \* audio/visual warnings
  - \* signal: 1 x RPM
- **Gem engine Cyclic Life Control**
  - \* Individual Aircraft Tracking
  - \* Rolls-Royce Service Bulletin
  - \* signals: 4 x RPM ( $N_{h1,2} + N_{f1,2}$ )
- **Usage Monitoring**
  - \* CLUMS data-base
  - \* lifting frame, sponson
  - \* signals:
    - 2 discretes (Wow, Radalt)
    - 15 analogue (alt, speed,  $\phi, Nr, Nh, Nf, Tq$ , Sponson strains, spares)



# Fatigue Life Monitoring - Now -

---



# Fatigue Life Monitoring -Transition-

Off-line  
processing

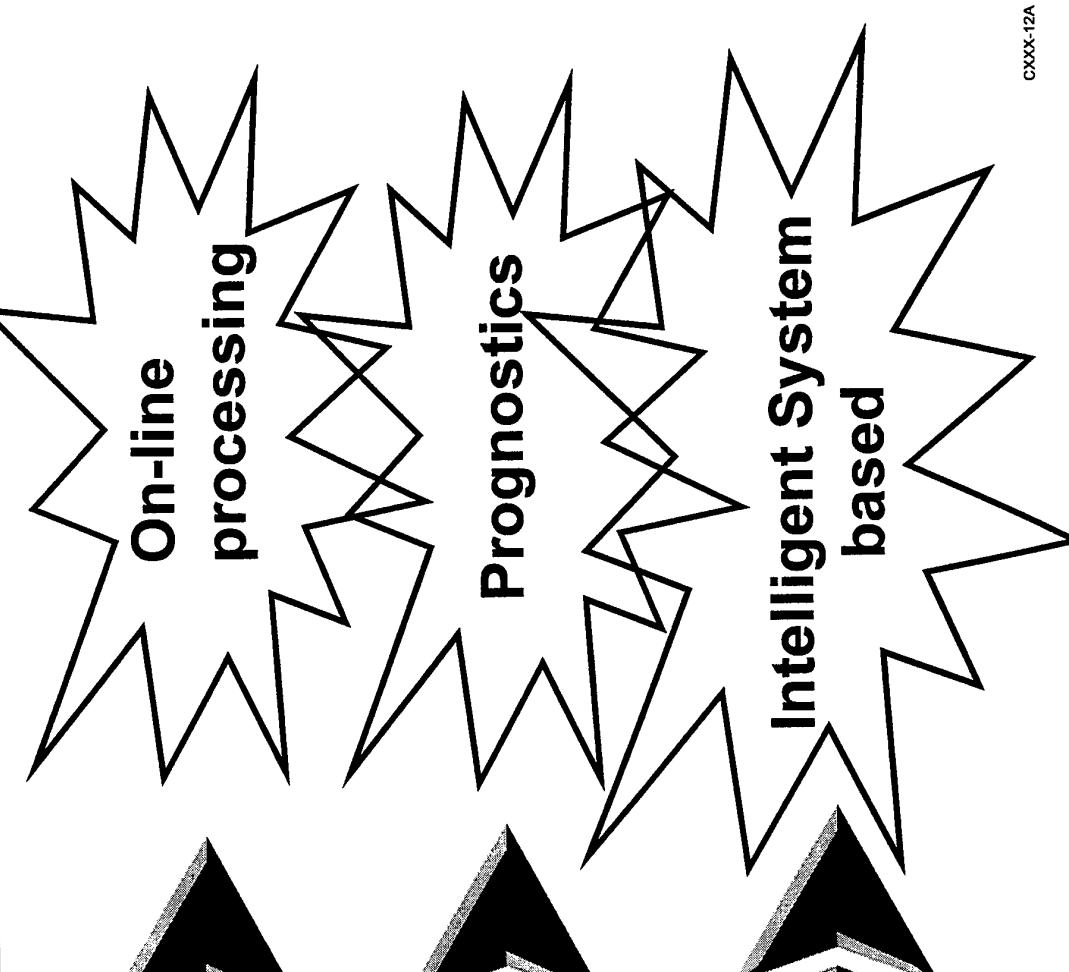
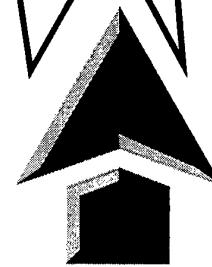
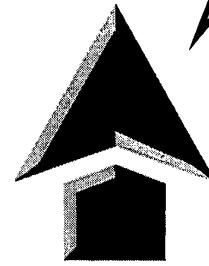
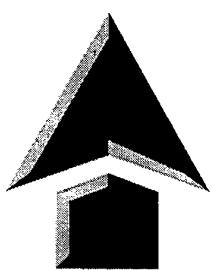
Diagnostics

Sensor based

On-line  
processing

Prognostics

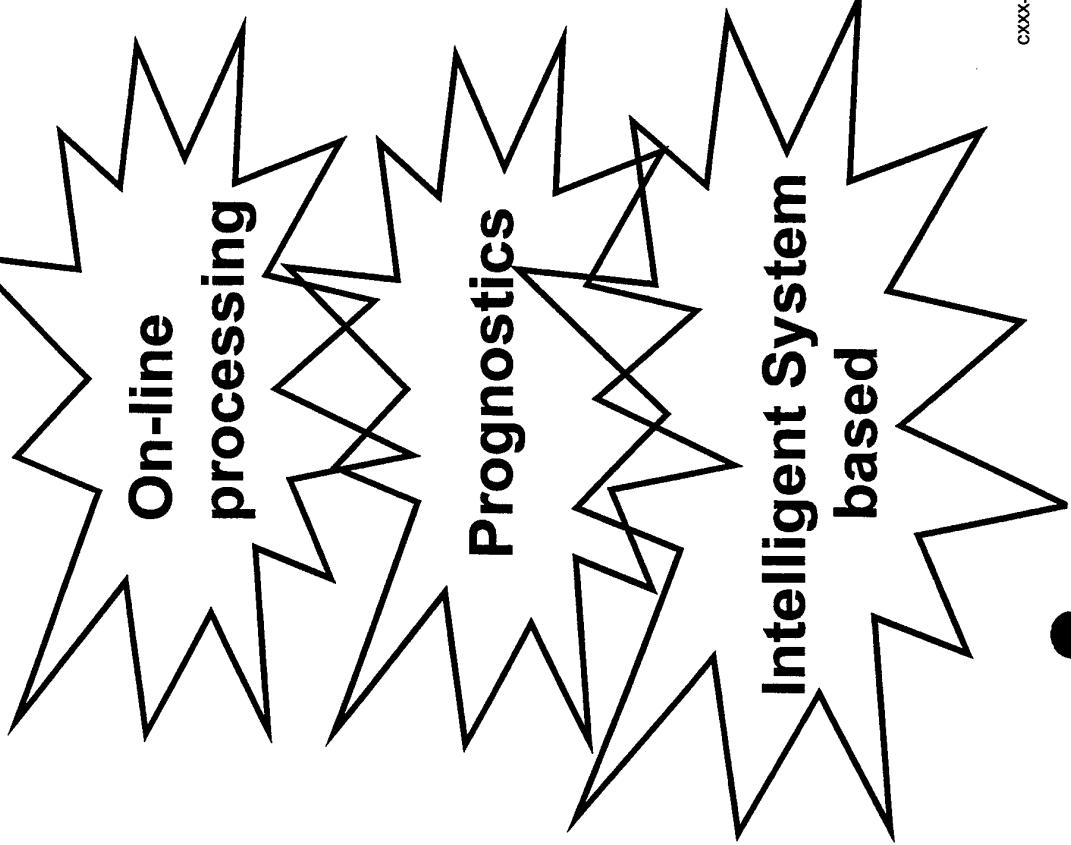
Intelligent System  
based



# Fatigue Life Monitoring - Future -

## Prognostics and Health Management (PHM) systems

- Intelligent Load Monitoring ILM
  - virtual strain sensors
  - neural networks
- Data Mining
  - autom. regime recognition
  - engine prognostics
- Reasoners
  - model, rule, case based
  - genetic algorithms



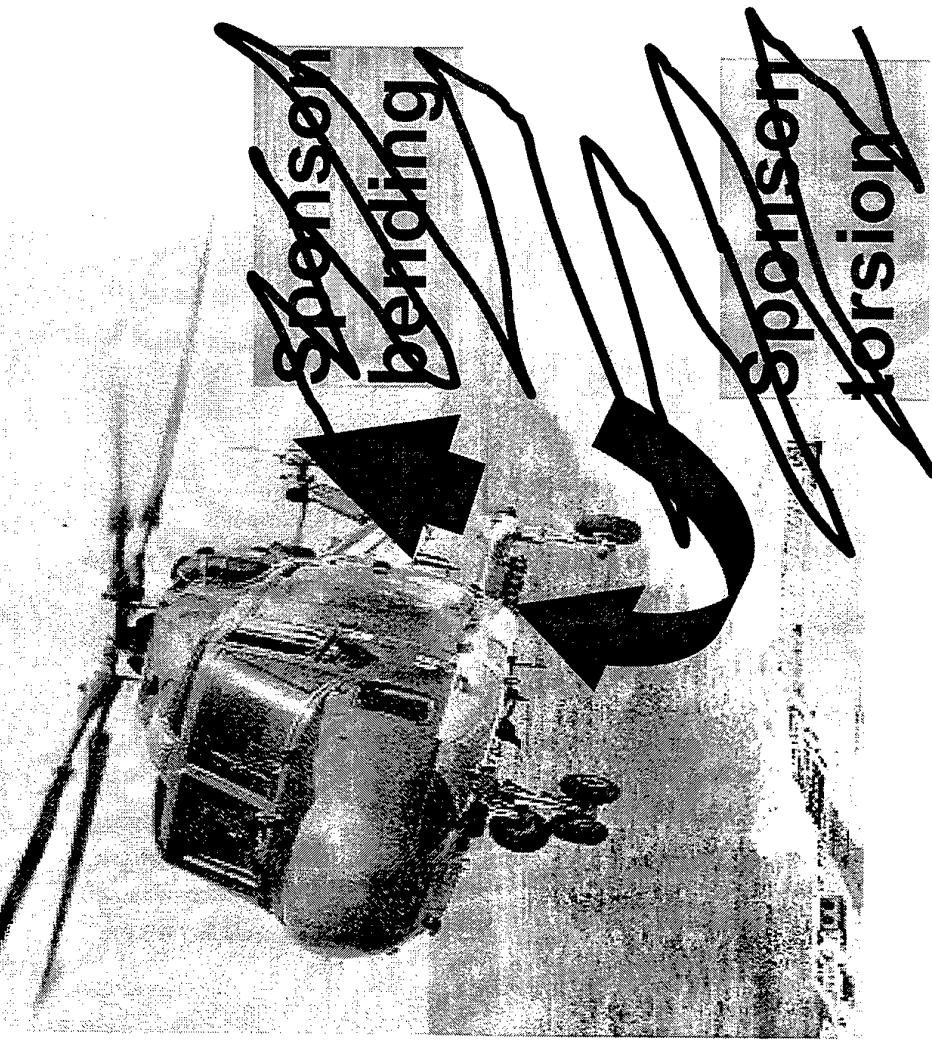
# ILM example - Now -



## Off-line Assessment of:

- Landing Type
  - deck/land
  - harpoon Y/N
- Landing Regime
  - running
  - MPOG
  - T.O. and Landing Weight statistics
- Limit Load Exceed.
- Vibration Level
- Damage accumulation

# ILM example - Future -



## PHM will enable:

- Elimination of strain sensors
- Strains derived from configuration, flight and environmental data
- On-line calculation of accumulated damage
- Prognostics for fleet and mission planning

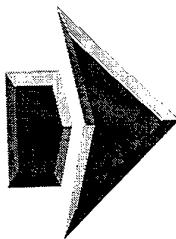
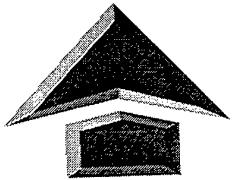
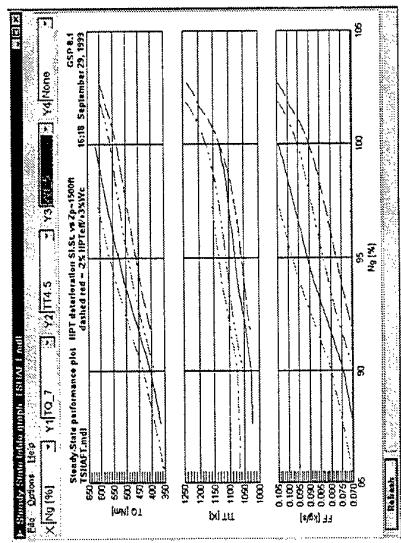
# Engine prognostics

## Problem: HPT Deterioration

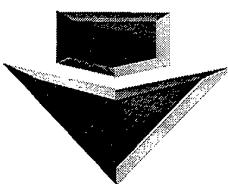
**Sea-Level efficiency - 2% with  
massflow +3%**

**Deterioration effect is hidden  
due to flight condition  
changes (causing variation  
in performance data)**

## GSP Turboshaft model



**Prediction of  
Deterioration Effects**



**Data Mining finds Known  
AND Unknown relations**

# Finally

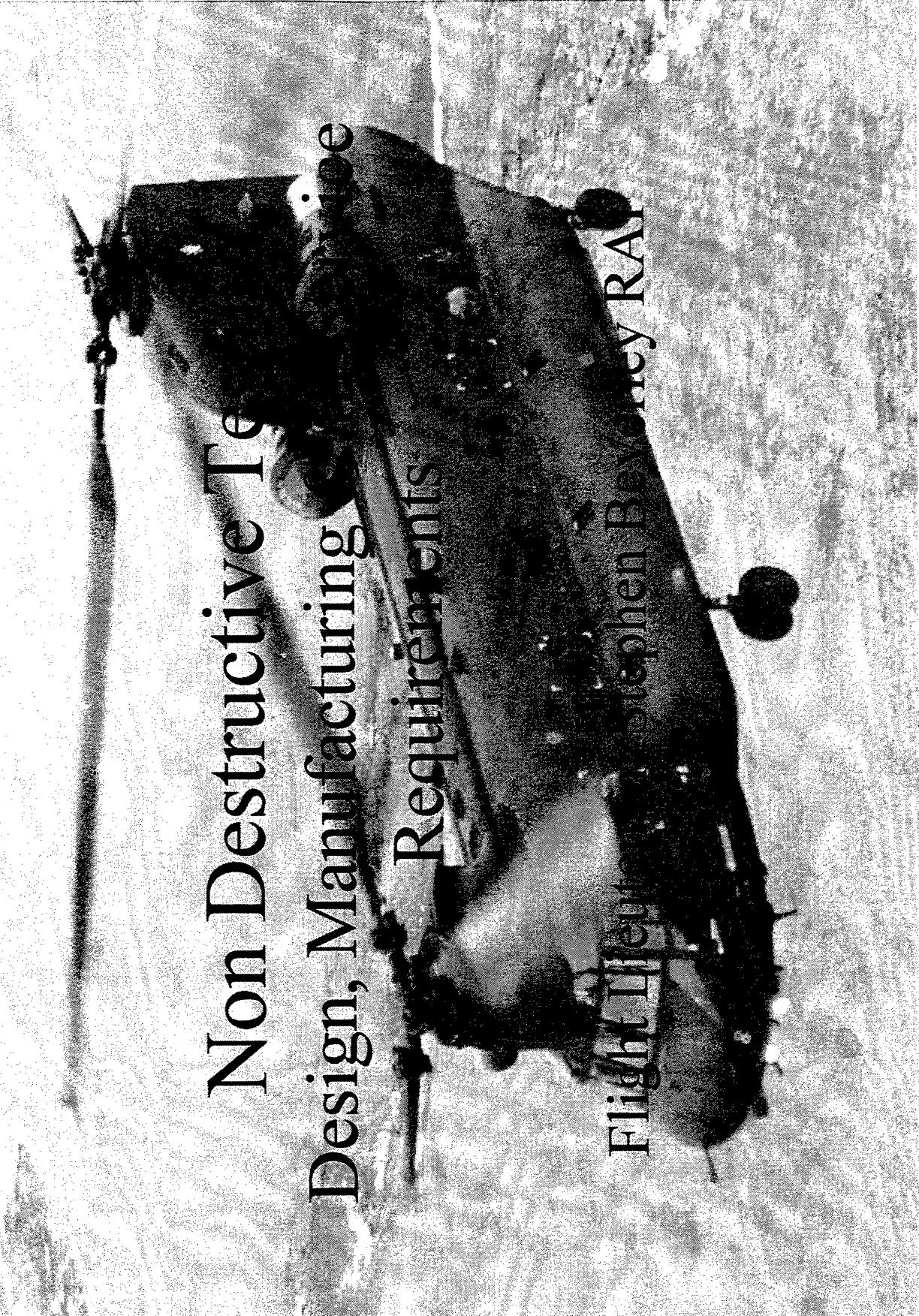
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- Better understand 20th century technology as a basis for future developments
- 21st century goal: Transition of emerging fixed-wing PHM technologies to the rotary-wing environment
- R&D Structural Integrity \$\$ will shift from the traditional

*Mechanical and Materials Engineering*

to

*IT, Quality Assurance and Certification areas*



# Non Destructive Testing Design, Manufacturing Requirements

Flight Test

Stephen B. Tracy RA

*“Human history becomes more and more a race between education and catastrophe”*

H.G. Wells

The Outline of History

*“Despite the acclaimed solid education of engineers, it is my experience in teaching fracture mechanics...to practicing engineers, that most have only a vague idea of such subjects as plastic deformation and design; to many Mohr's circle is an enigma; at most one in a class knows the stress concentration factor of a circular hole; fewer even remember yield criteria and their significance”*

David Broek

The Practical Use of Fracture Mechanics

# Scope

- Non-Destructive Testing
  - Design
  - Manufacturing
  - In Service
- Technology capabilities
- New Technology
- Reliability of Inspection
- Case Studies
  - Sea King
  - Puma

# NDT Methods

- Visual Aids
  - Magnifying Glass
  - Endoscopy (Rigid, flexible, CCD imaging)
- Dye Penetrants
  - Colour Contrast & Fluorescent
- Magnetic Particle
  - Colour Contrast & Fluorescent
  - Magnetic Field, Current AC/DC
- Eddy Current
  - Single/Dual/Multiple Frequency
  - Rotary (bolt hole)
  - Transient
  - AC/PD
- Ultrasonics
  - Compression, Shear, Lamb wave
  - Spectroscopy
  - Air Coupled
  - Low frequency techniques (Resonance, tap testing)
- Acoustic Emission
  - Radiography
  - X - γ Ray
  - Filmless (Digital imaging)
    - Phosphor screen
    - Amorphous Selenium
- Optical Techniques
  - D-Sight
  - Edge of Light

# NDT Requirements

- Design
  - Determine SSIs for design requirements (Fail Safe, Safe Life or Damage Tolerance)
  - Stress analysis (operational, overload,...)
  - Life component based on Fracture Mechanics, expected operational loads & environmental factors
- Determine appropriate NDT technique to find defects or faults in structure

# NDT Requirements

- Manufacturing
  - Process/Quality control to filter components exceeding acceptable defect criteria
  - Inspection environment enables high quality processes.
- Automated: PFD lines, Ultrasonic & Eddy Current C-scanning.
  - Scope to develop & implement new technologies
- Negative aspect: NDT is carried out pre-assembly (potential for induced defects)

# NDT Requirements

- In-Service
  - Operator Needs:
    - Cheap/Cost effective
    - High reliability (no false calls)
    - Quick
    - Low frequency of inspection
  - Design for Inspection
    - Accessibility
    - Material selection & manufacturing processes compatible with in-field inspection capabilities
      - Increase Inspection reliability
      - Reduces time & maintenance cost

# New Technology

- Aim of implementing new technology is to improve inspection reliability and efficiency, and reduce costs through automation
  - Pulsed Thermography
  - Laser Shearography
  - Field capable C-scan systems
  - Digital X-ray Techniques
- New analyses can lead to unexpected results
  - Complex image interpretation (automated or manual) can lead to increases in false calls
- New Technology has to provide significant improvement in capability over current practices
  - Simplicity
  - Data interpretation
  - FAA & USAF development programmes are sustaining impetus on technology developments from the private sector

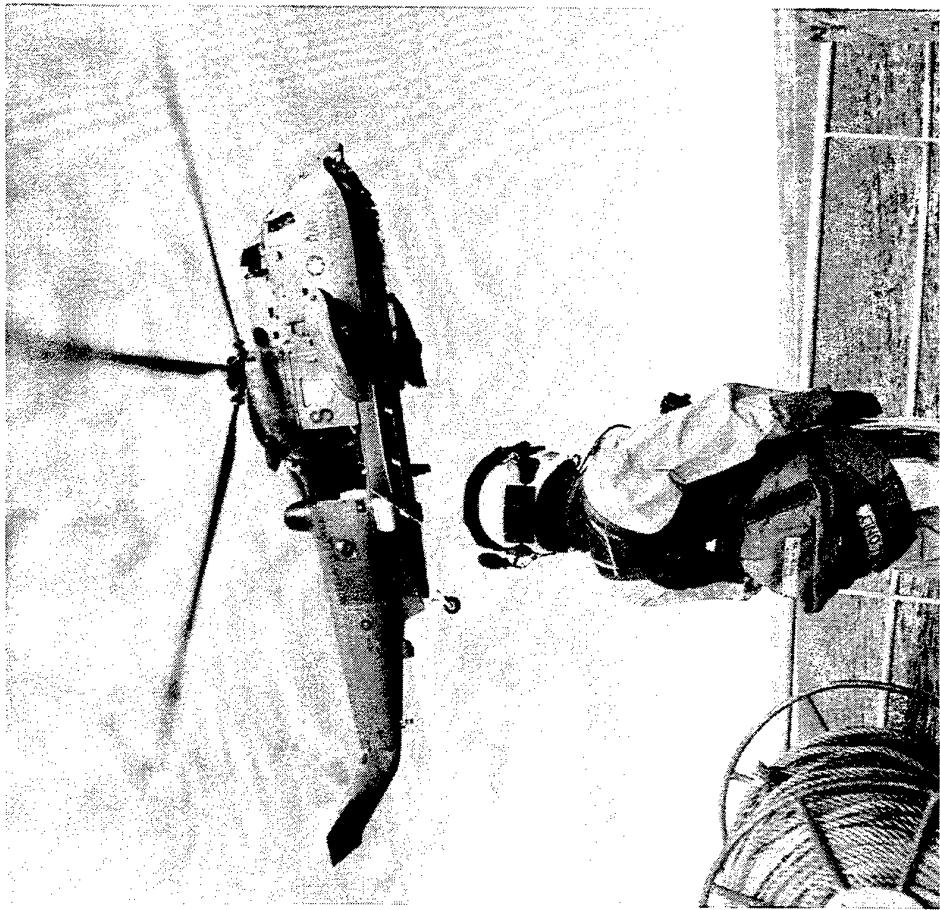
# Reliability of Inspection

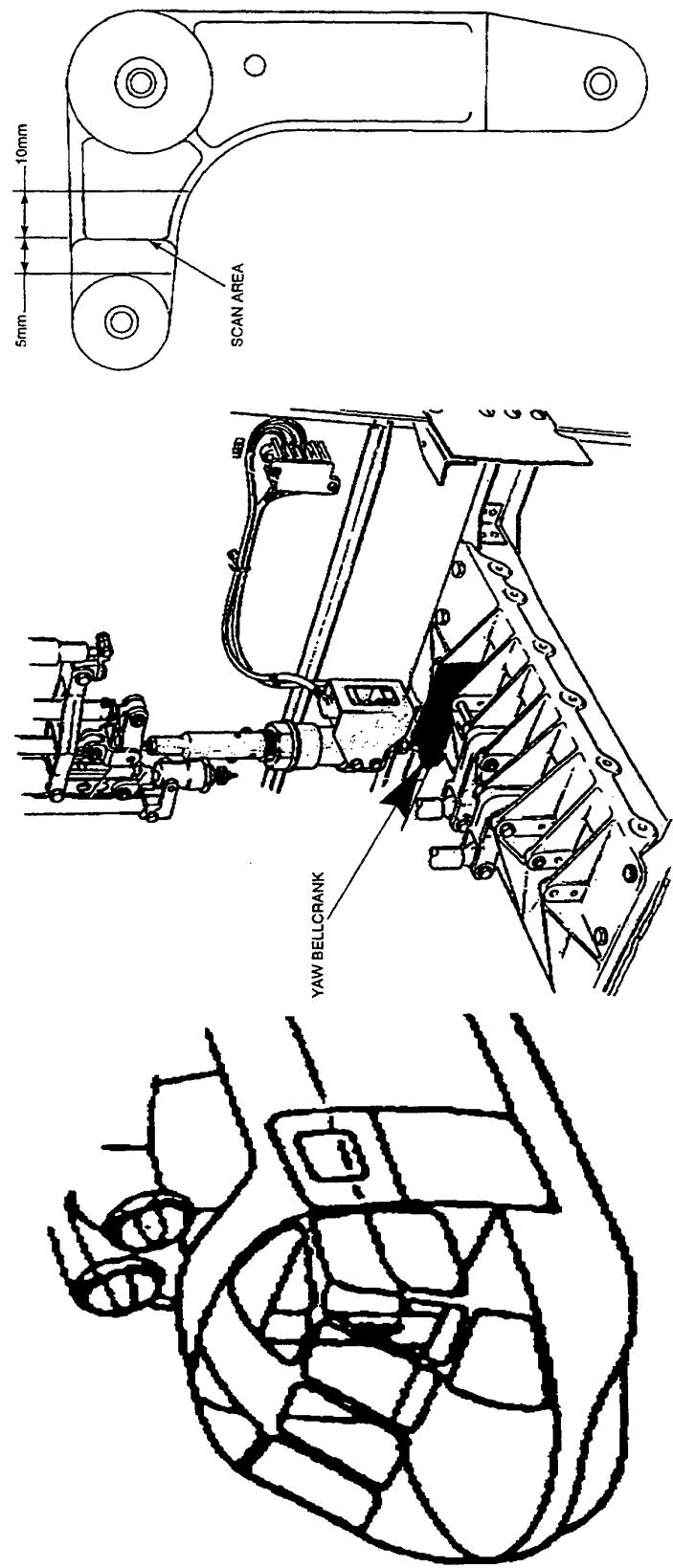
- $R = f(\text{technique, method, human factor})$
- Inspection points must be derived to optimise the selection of technique
- Matching inspections with appropriate level of maintenance
  - Incremental increase in risk!
- Experience derives the most effective technique, and associated limitations
- Ultimately, no system can ever be 100% perfect

# NDT Challenges

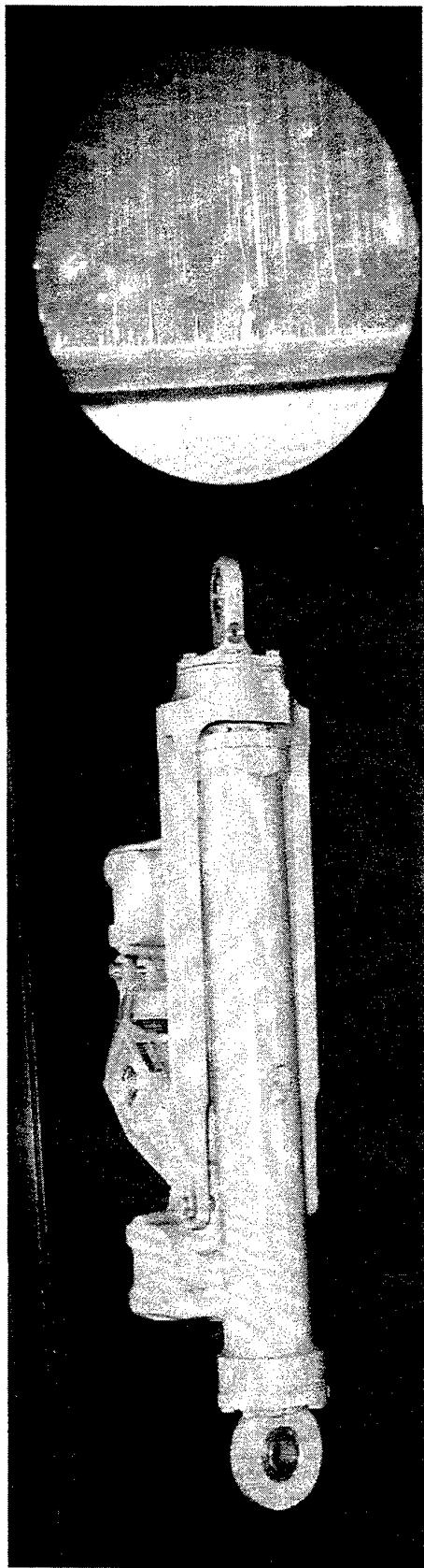
- What if the initial flaw size is below the threshold of detection?
- Durability approach, where no inspections can or will be done.
  - Retire the component after H/2 hrs
  - Expensive
- Proof Testing
  - Apply every Safe operation period ( $a_{\text{proof}}$  to  $a_p$ )
  - Safe operation period can be short leading to high maintenance penalty
- Stripping
  - Machining away a surface layer equivalent to  $a_p$
  - Applicable to fillet radii or fastener holes that require oversizing

# Case Study 1: Seaking





## Case Study 2: Puma MRH Servo



# Summary

- NDT is a limiting factor in Damage Tolerant Design
- Pragmatic use of inspection capabilities can facilitate effective fatigue life management
- New technologies must be thoroughly evaluated before implementation either for Quality Control or in-service inspections
- Other inspection/maintenance methods may be required to validate structural integrity of rotorcraft if Damage Tolerance is rigorously applied

# Usage monitoring and materials factors in achievement of damage tolerance

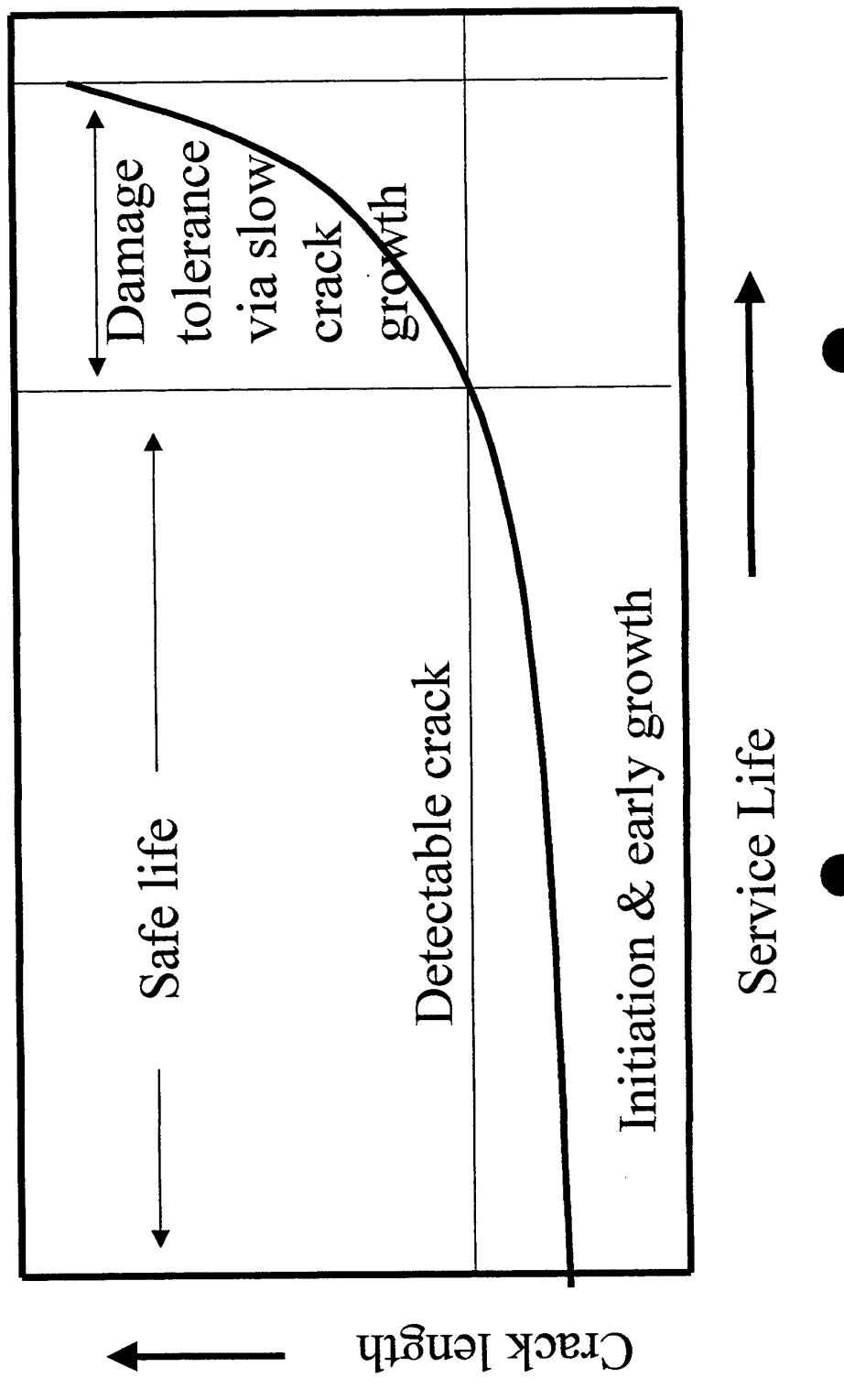
Phil Irving, R A Hudson  
Cranfield University

Damage Tolerance discussion workshop  
4-5 April 2000

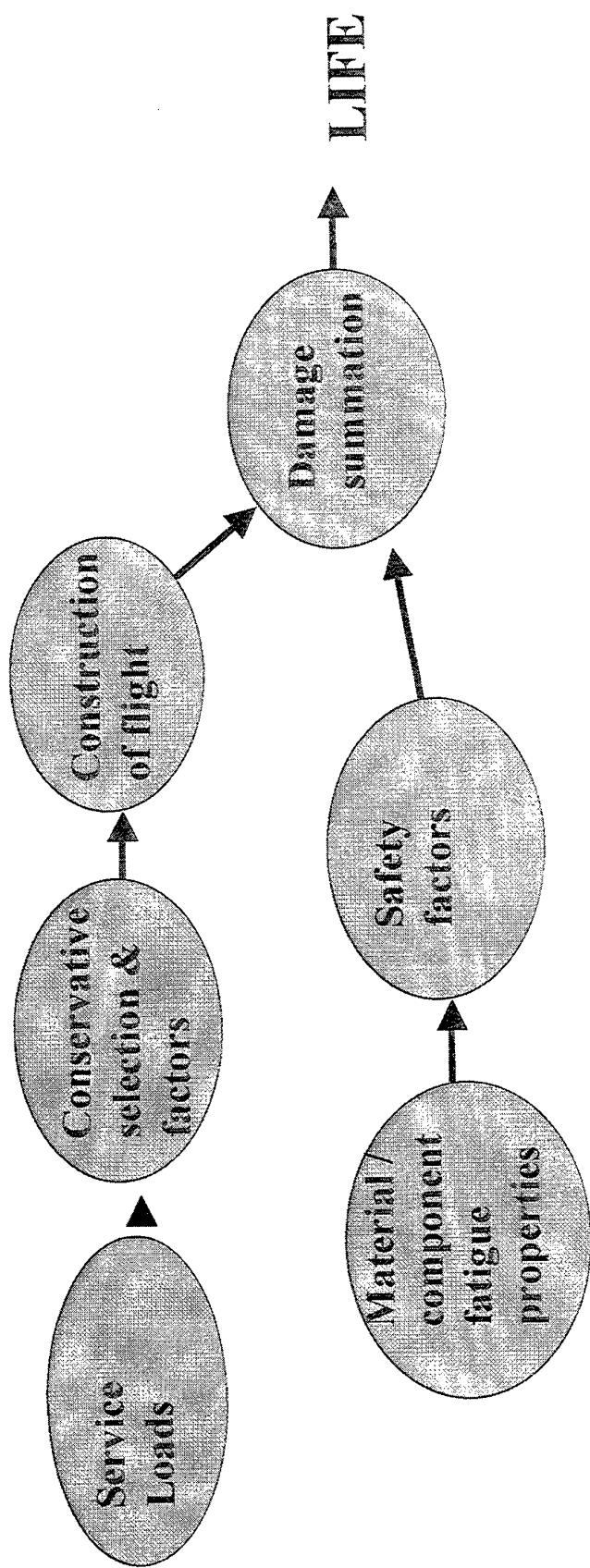
## Objectives

- Determine benefits of Usage monitoring in achievement of damage tolerance in helicopter components
- Establish how much difference material choice makes to the damage tolerant situation

## Damage Tolerance- schematic



## Safe life calculation- schematic



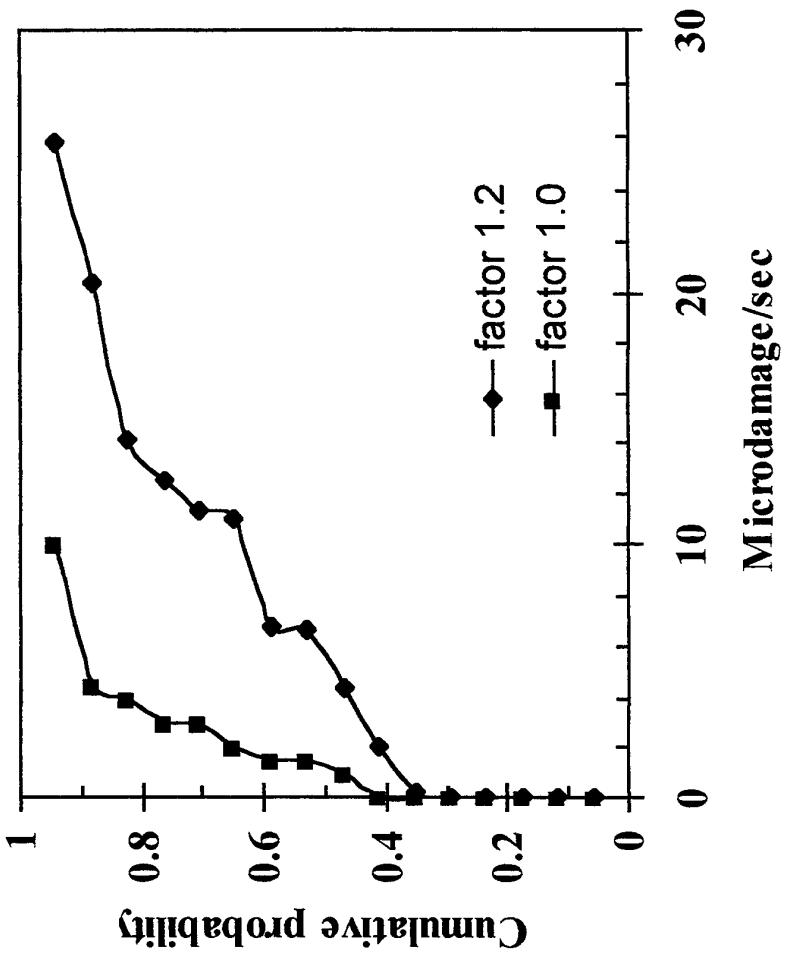
## Work- outline

- 10 hours of service load data containing all manoeuvres in design spectrum for Lynx; select dogbone rotor component for study
- Extract load histories for up to 20 examples of each manoeuvre in design spectrum
- Calculate damage for each manoeuvre using component fatigue data supplied by manufacturer
- Develop worst case design spectrum for deterministic life
- Compare worst case fixed lives with distributions of lives developed using Monte Carlo simulations

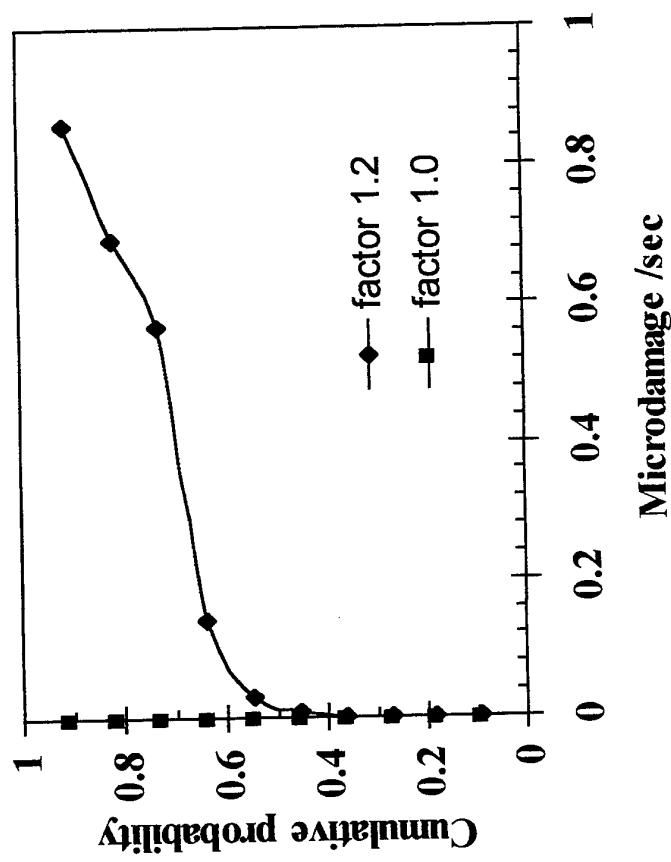
## Work outline - continued

- Repeat for crack growth lives from 1.27 mm defect
- Repeat for alternative materials- steels, aluminium alloys

# Variability in manoeuvre damage-control reversal



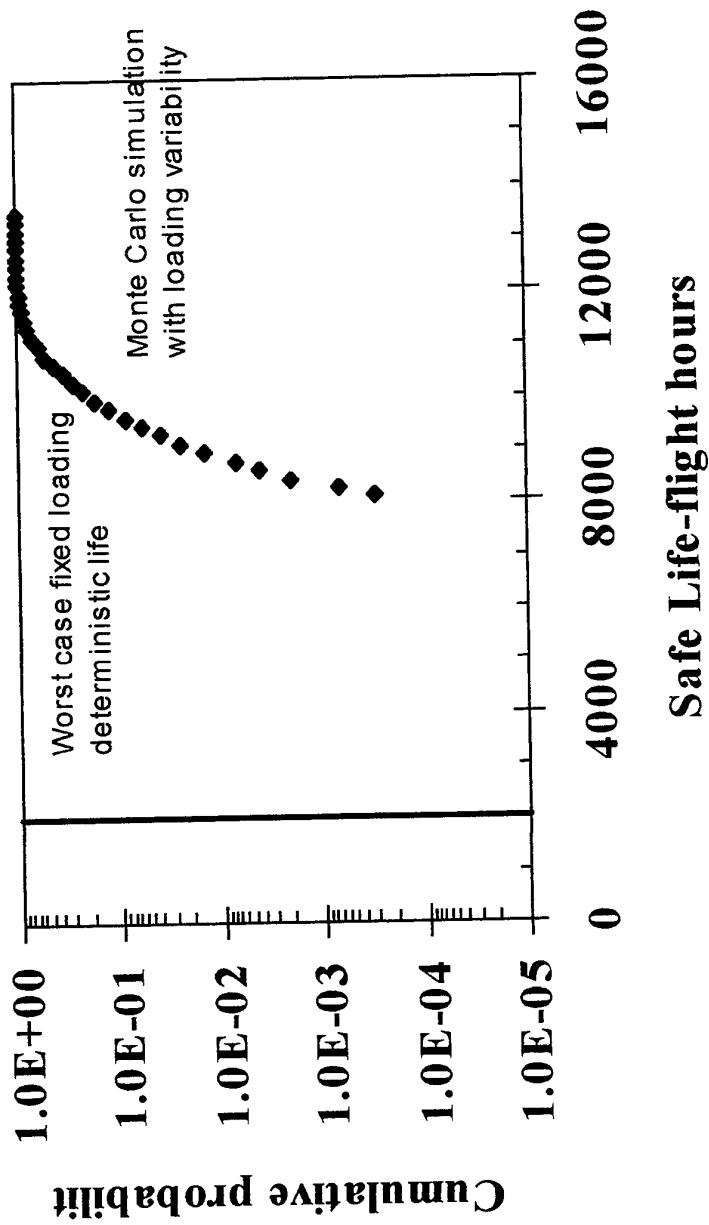
# Variability in manoeuvre damage- Hover



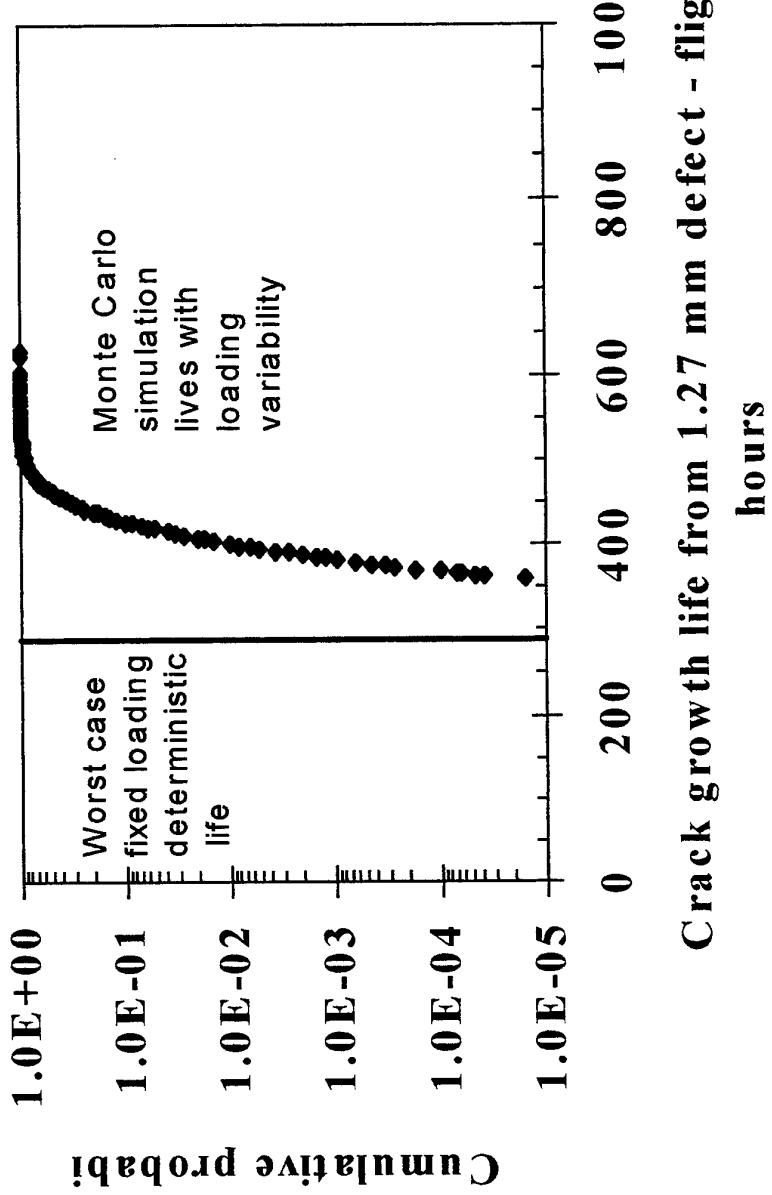
## Safe life & crack growth comparison- flight hours

| Fatigue design approach | Worst case | Simple worst case | 50% probability | Simple 50% |
|-------------------------|------------|-------------------|-----------------|------------|
| Safe life               | 2,020      | 273               | 27,778          | 2,817      |
| Crack growth            | 290        | 73                | 636             | 183        |

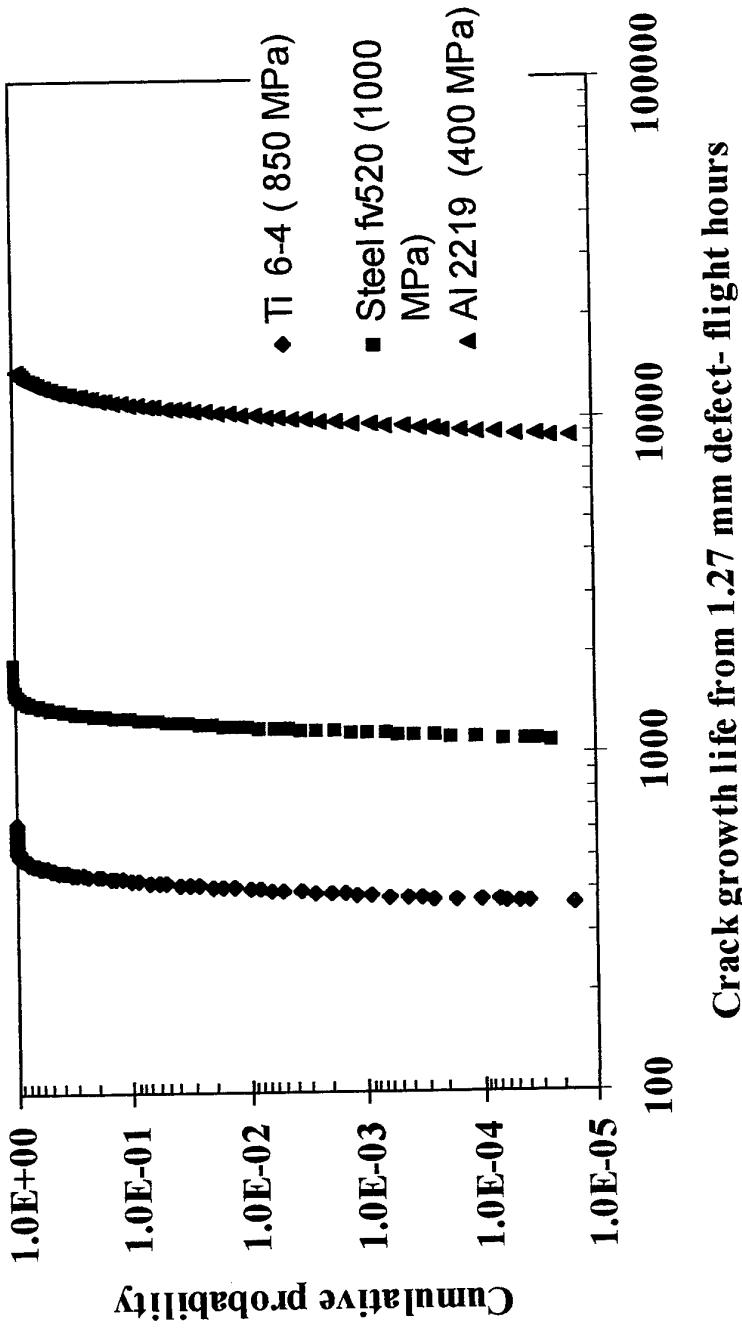
# Monte Carlo simulations- safe life-titanium



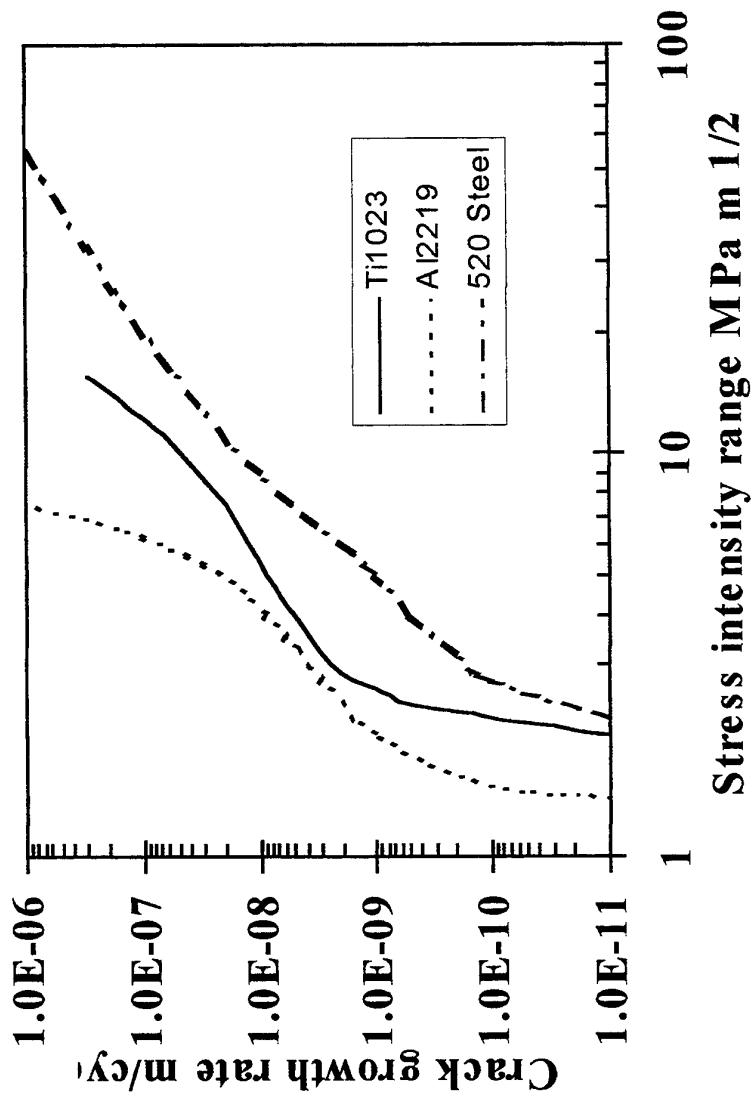
# Monte Carlo simulations- crack growth-titanium



# Monte Carlo simulations- crack growth - effect of material- stresses normalised for strength



# Crack growth data



## Conclusions

- Usage monitoring could improve damage tolerance capability in helicopter rotor components by elimination of uncertainties in service loading;
- Inspections when calculated crack lengths indicate requirement
- Lives to grow cracks from 1.25 mm to 18 mm increased from 290 hours for worst case loading to a most probable life of 500 hours
- Both steel and aluminium alloys show greatly increased crack growth lives when loads normalised for strength

## Acknowledgements

Thanks are due to many colleagues at Cranfield, CAA  
and GKN WHL for helpful discussions.

CAA is thanked for continuing support for the CAA  
chair.

## **SESSION D**

# **AIRFRAME STATIC COMPONENTS**



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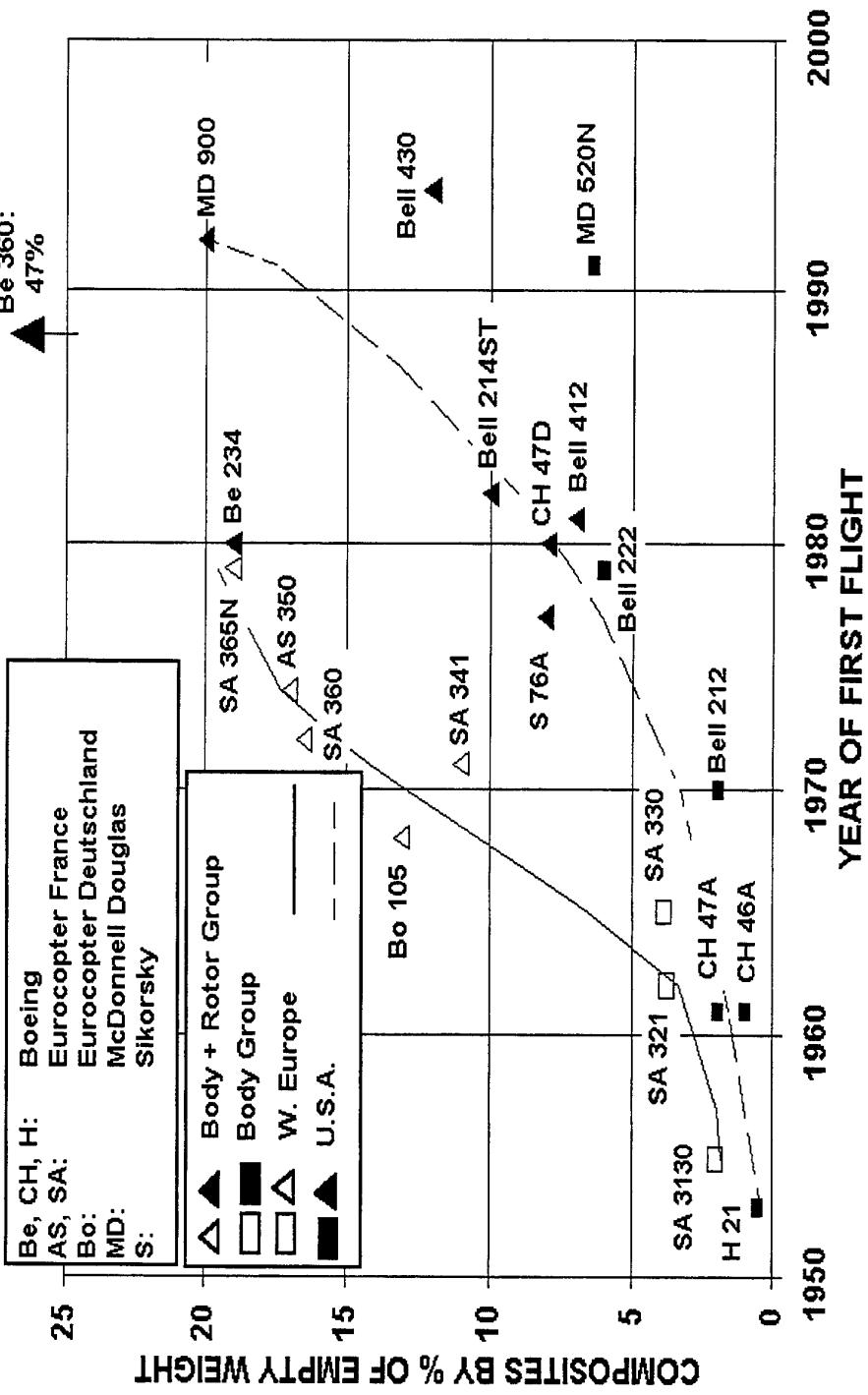


# FAILURE PREDICTION OF COMPOSITE FUSELAGE STRUCTURES

Joyanto K. Sen and Jian Li  
The Boeing Company  
Mesa, Arizona, USA

Workshop on Damage Tolerance in Helicopters  
Cranfield University, UK  
4-5 April 2000

# INCREASING USE OF COMPOSITES IN HELICOPTERS



# DESIGNING COMPOSITE STRUCTURES

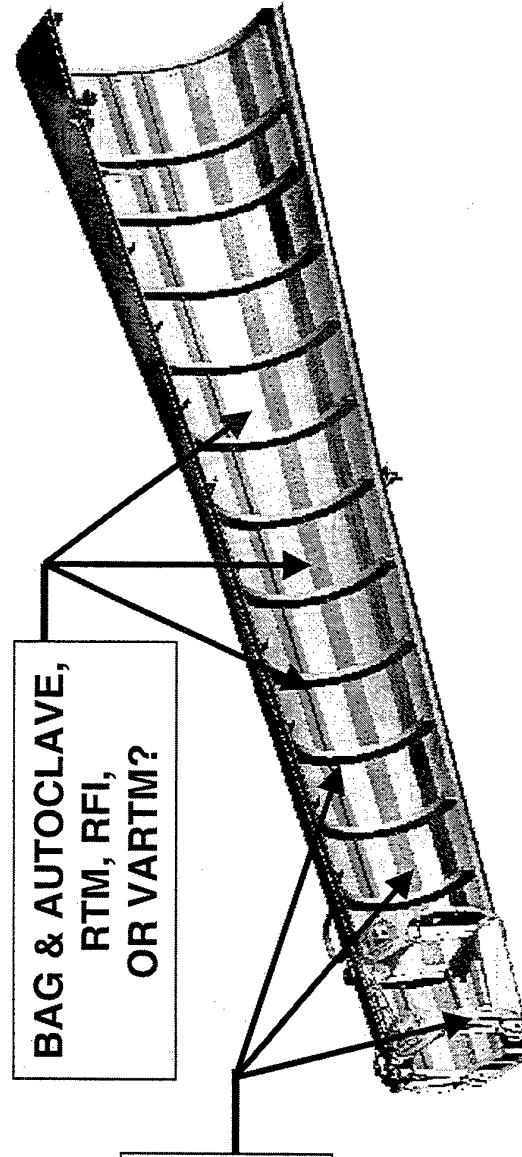


- EXHIBITS MULTIPLE FAILURE MODES
  - COMPOSITES ARE MADE OF MULTIPLE MATERIALS
  - MECHANICAL BEHAVIOR CAN BE TAILORED
- DESIGN IS VALIDATED BASED ON
  - MATERIAL CHARACTERIZATION
  - STRUCTURAL ANALYSIS
  - VERIFICATION THROUGH TEST
- LIGHTWEIGHT, AFFORDABLE DESIGNS DEPEND ON SEVERAL DESIGN AND MANUFACTURING ITERATIONS

# TYPICAL STRUCTURES REQUIRE INNOVATIONS FOR LIGHTWEIGHT, AFFORDABLE DESIGNS



- HAND LAID-UP PREPREG, OR DRY FIBER WITH LIQUID MOLDING?
- UNI-DIRECTIONAL TOWS, OR CLOTH, OR OTHER FABRIC FORMS?
- STITCHED, OR OTHER METHODS OF ENHANCED SURVIVABILITY AND IMPROVED HANDLING CHARACTERISTICS
- AUTOCLAVE-CURED, OR THERMOFORMED, OR SELF-CONTAINED TOOL
- THERMOSET, OR THERMOPLASTIC?



# COMPLEXITIES OF COMPOSITE DESIGNS



- DESIGN ITERATIONS INFLUENCE MATERIAL BEHAVIOR
  - MATERIAL & CORE FORMS
    - PLY ORIENTATIONS
    - RESIN SYSTEMS
      - STACK-UP SEQUENCE
  - MANUFACTURING PROCESS INFLUENCES VALIDITY AND NUMBER OF ITERATIONS OF ANALYSES
    - RESIN POCKETS
    - CO-BONDING & CO-CURING
    - MATERIAL FORMABILITY
      - FIBER VOLUME
      - PLY WAVINESS
      - LAMINATE QUALITY
    - MISPLACED PLY TERMINATION & PARTS LOCATIONS

# REQUIREMENTS OF INDUSTRY

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- REDUCE MATERIAL CHARACTERIZATION CYCLE TIME
  - STATIC
  - FATIGUE
- EASY ITERATION FOR MATERIAL CHOICES AND INSERTION
  - RAPID DEVELOPMENT OF MATERIAL PROPERTIES
- REDUCE DESIGN CYCLE
  - SIMPLIFIED AND RAPID ANALYSES FOR STRUCTURAL INTEGRITY: FAILURE MODES AND LOCATIONS, AND MANUFACTURING DEFECTS

# APPROACH TO COMPOSITE ANALYSIS?

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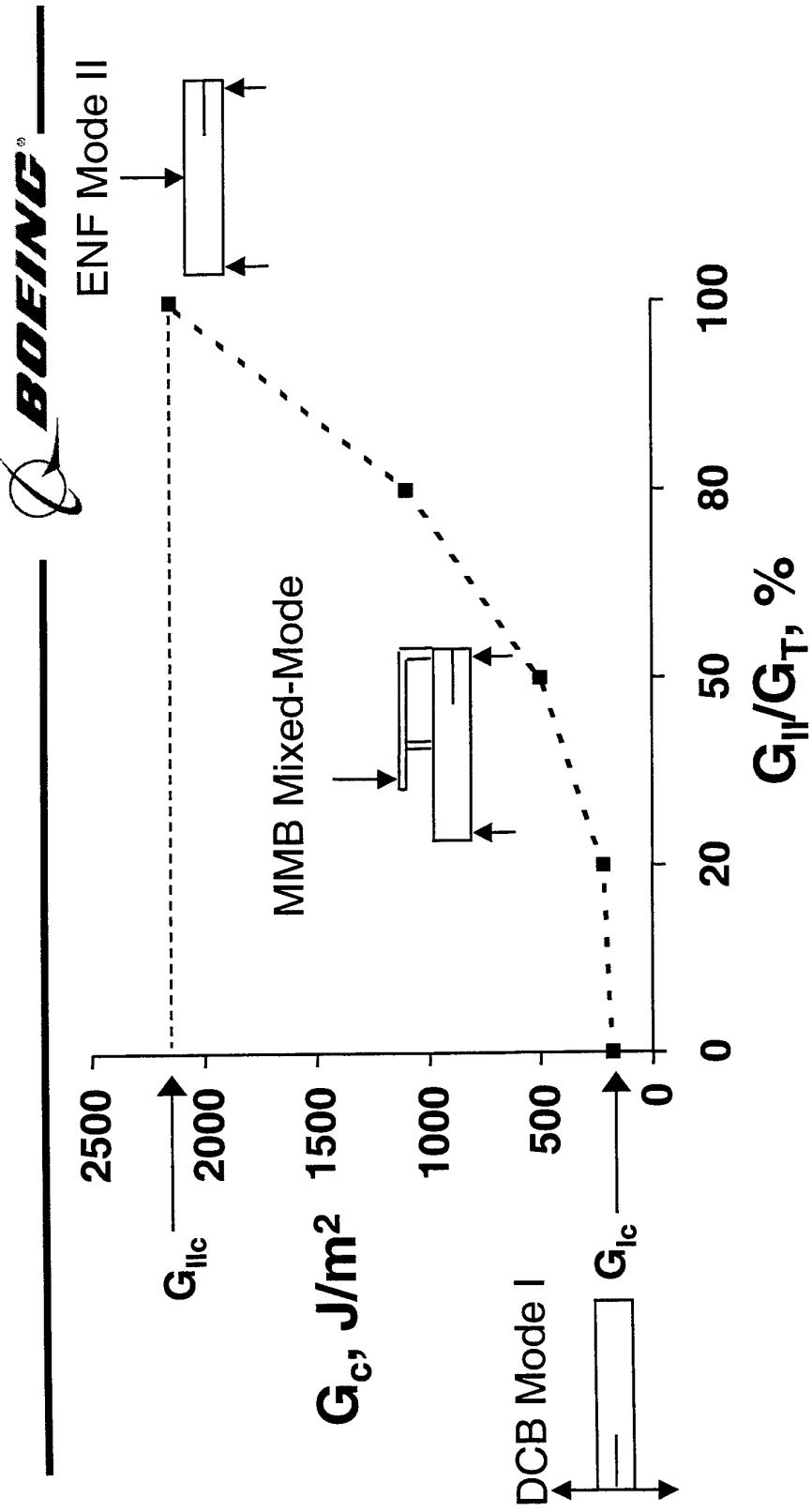
- **STRUCTURAL ANALYSIS BASED ON FAILURE MODE(S)**
  - REDUCES MATERIAL CHARACTERIZATION CYCLE TIME
  - PREDICTED STRENGTH VALIDATED FOR CRITICAL LOCATION
  - IDENTIFIES INSPECTION TIMES BASED ON FAILURE-CRITICAL LOCATIONS
- **PARAMETRIC MODELING**
  - PARAMETRIC CHANGES OF THE STRUCTURE ARE RAPIDLY MODELED TO SIMULATE MANUFACTURING DEFECTS FOR STRUCTURAL ANALYSIS
  - REDUCES ANALYSIS CYCLE TIME

# FRACTURE MECHANICS APPROACH TO FAILURE PREDICTION METHODOLOGY



- ASSUME CRACK AND DELAMINATION
  - DELAMINATION IS PREDOMINANT FAILURE MODE IN COMPOSITE STRUCTURES
- COUPON TESTS TO DETERMINE MIXED-MODE MATERIAL FRACTURE TOUGHNESS ( $G_C$ )
- FINITE ELEMENT ANALYSIS TO CALCULATE TOTAL STRAIN ENERGY RELEASE RATE ( $G_T$ ) AND MIXED MODE RATIO ( $G_{II}/G_T$ )
  - VIRTUAL CRACK CLOSURE TECHNIQUE

# MIXED-MODE DELAMINATION CRITERION FOR A GIVEN MATERIAL SYSTEM



$$G_c = M_0 + M_1 (G_{II}/G_T) + M_2 (G_{II}/G_T)^2 + M_3 (G_{II}/G_T)^3$$

## FAILURE LOAD PREDICTION



- FAILURE OCCURS WHEN

$$[G_T(G_{II}/G_T)]_{FE\text{ ANALYSIS}} = [G_C(G_{II}/G_T)]_{TEST}$$

- FROM  $G_{II}/G_T$  DETERMINE MIXED-MODE DELAMINATION FRACTURE TOUGHNESS,  $G_C$ ,
- FROM FE ANALYSIS CALCULATE CONSTANT OF PROPORTIONALITY,  $\lambda$ , FOR LOAD,  $P$ , IN PULL-OFF TESTS
$$G_T = \lambda P^2$$
- THE CRITICAL LOAD,  $P_C$ , AT DELAMINATION IS
$$P_C = \sqrt{(G_C/\lambda)}$$

## SUMMARY

---



- FRACTURE TOUGHNESS APPROACH IS PROMISING
  - REDUCES ANALYTICAL CYCLE TIME
  - REDUCES DESIGN DEVELOPMENT TIME
  - REDUCES COST
- FRACTURE TOUGHNESS APPROACH IDENTIFIES CRITICAL LOADS AND LOCATIONS, AND RAPIDLY ANALYZES SEVERAL MANUFACTURING DEFECTS
- FURTHER VALIDATED DEVELOPMENT OF FRACTURE TOUGHNESS APPROACH FOR WIDER STRUCTURAL APPLICATION IS NECESSARY

## LESSONS LEARNT FROM RECENT ATTEMPTS AT APPLYING DAMAGE TOLERANCE TO HELICOPTER AIRFRAMES

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HEAD OF STRUCTURES AND MATERIALS

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SENIOR STRESS ENGINEER  
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UK

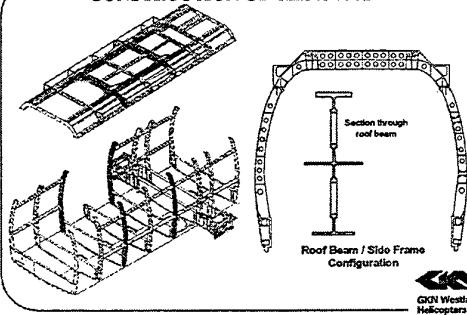


### PRESENTATION OUTLINE

- Introduction to the EH101
- Origin of damage tolerance requirement
- Overview of damage tolerance evaluation
- Test Programme
- Crack growth analysis & rotorcraft specific problems
- Results of the analysis
- Reasons for the results and results in perspective
- Alternative approaches
- Conclusions



### CONSTRUCTION OF THE EH101



### SUBSTANTIATION OF THE EH101

- EH101 Airframe **SAFE LIFE** demonstrated by a full-scale factored load fatigue test.
- Failure modes observed during this test were eliminated from the production standard by design changes. A stand-alone fatigue test of a production standard lift frame, and fine mesh finite element modelling, demonstrated the effectiveness of the changes.
- Safe Lives in excess of 10000 hours have been demonstrated for civil and military variants.
- Damage tolerance evaluation in accordance with JAR 29.571 still required by CAA/RAI.



### DAMAGE TOLERANCE ASSESSMENT OF THE EH101 MAIN LOAD PATH

#### Programme of testing initiated :

- material data
- structural elements
- full-scale components
- full-scale airframe test.

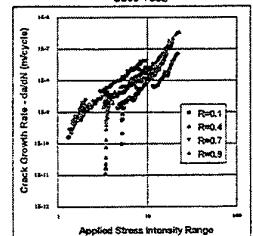
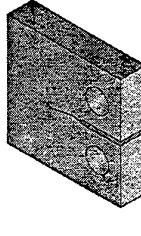
#### Objectives :

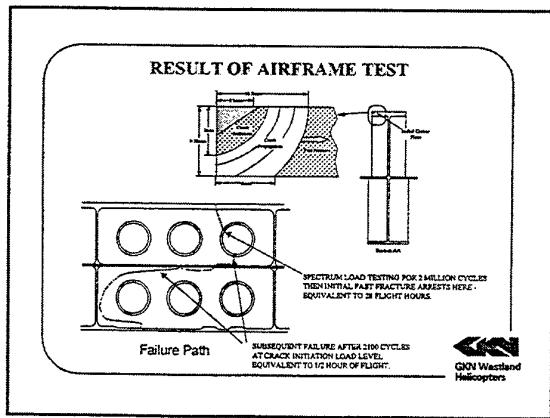
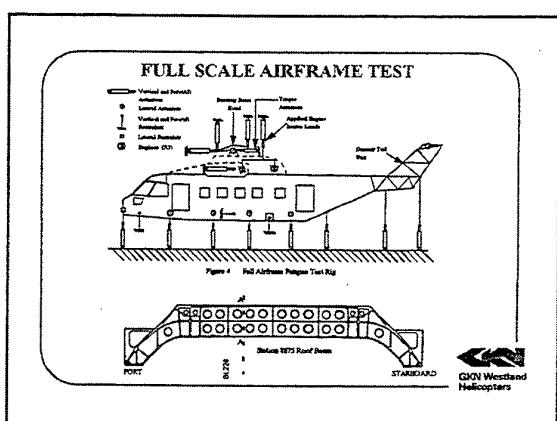
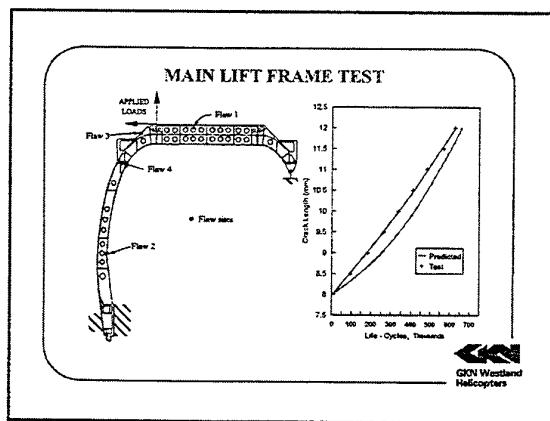
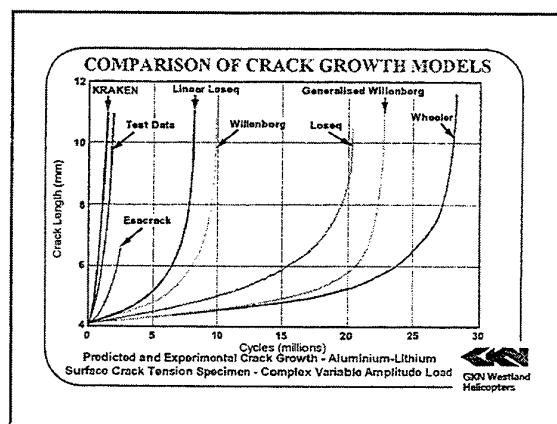
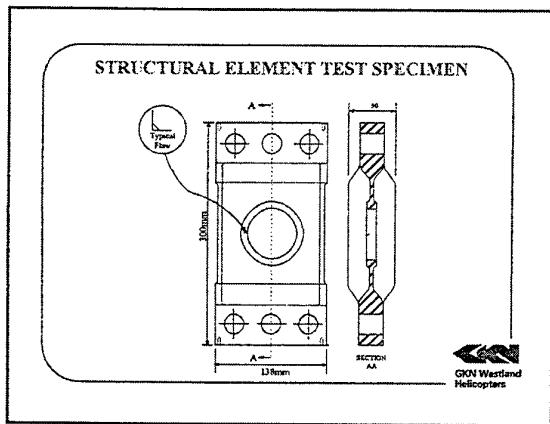
gather basic material data - develop and prove analytical models - generate compliance functions - demonstrate crack trajectories and failure modes.



### COMPACT C TEST SPECIMEN

Aluminum Lithium Alloy Forging  
8090-T852





**LESSONS LEARNT FROM TEST PROGRAMME**

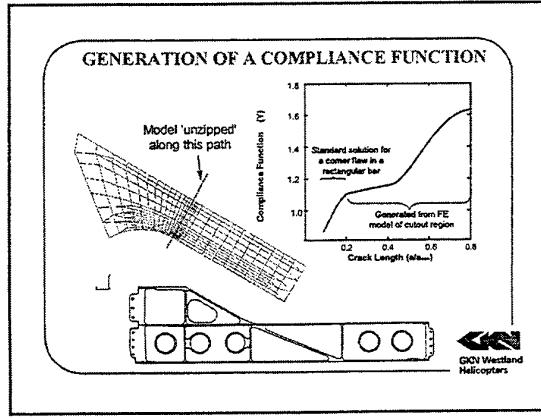
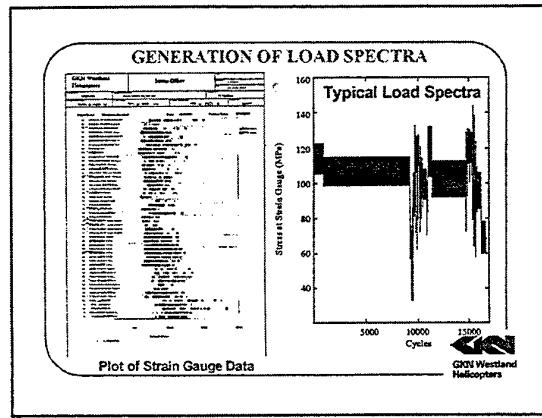
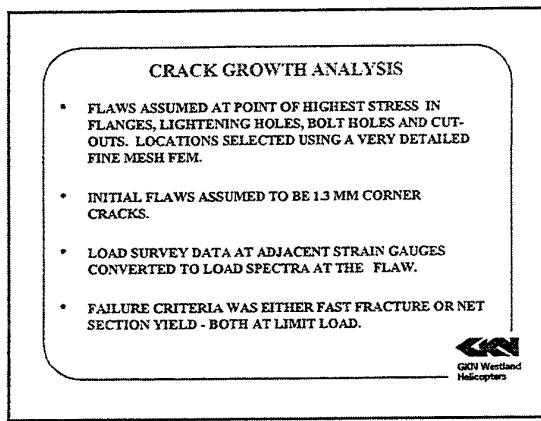
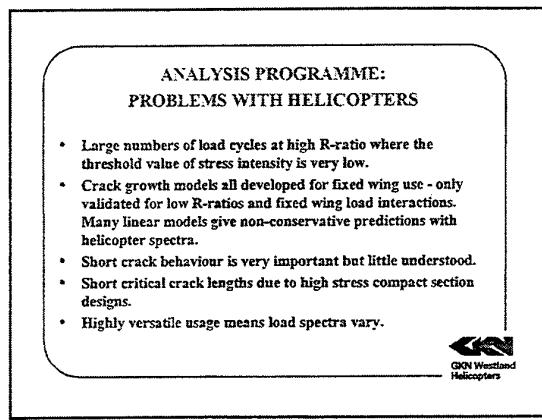
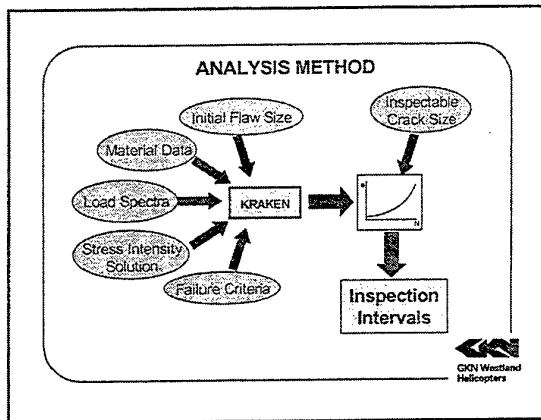
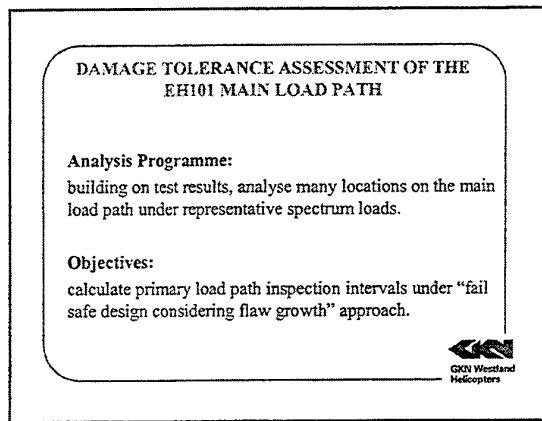
Cracks are very difficult to initiate - even from 4mm saw cuts, scalpel-sharpened and etched.

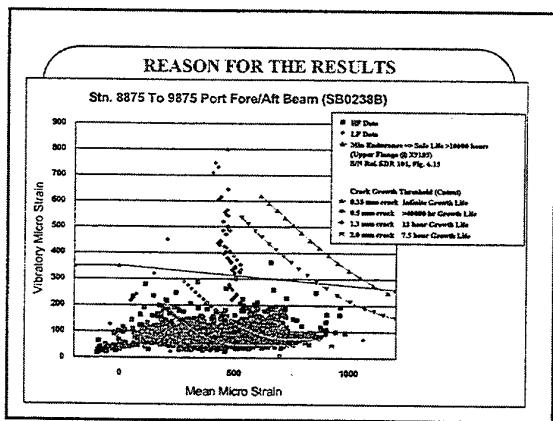
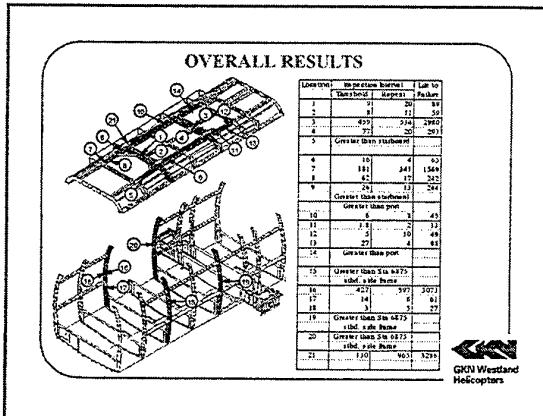
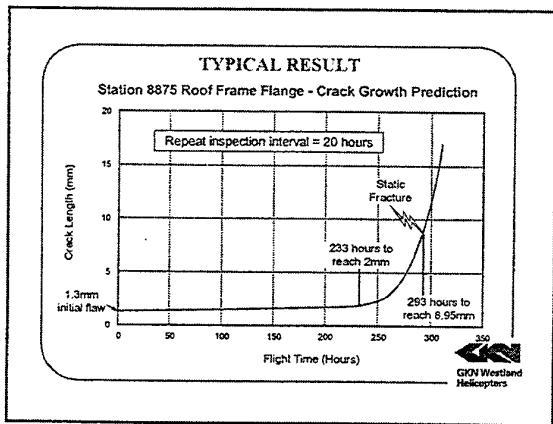
Critical crack lengths are short and no significant load redistribution occurs before fast fracture. Primary load paths are likely to fail completely.

Accuracy of crack growth predictions depend on accuracy of compliance functions.

Most analytical models are non-conservative under complex helicopter spectra loads (even linear summation models).

The GKN Westland Helicopters logo is at the bottom right.





**RESULTS IN PERSPECTIVE**

- Airframes do not crack regularly in service and result in catastrophic failure.
- These results are NOT related to Al-lith which, in fact, has superior da/dN curves to 7010 in the working range.
- Manufacture and husbandry are of a high standard.
- Flaws and damage are NOT cracks - cracks proved difficult to initiate on test from severe, etched saw cut flaws.
- For analysis the flaws are assumed at the points of highest stress - this is unlikely.
- Limit load events are very rare.

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**ALTERNATIVE APPROACH - SECONDARY LOAD PATH**

POOR RESULTS ABOVE REFER TO DETECTING CRACKS IN THE PRIMARY LOAD PATH BEFORE IT FAILS.

**ALTERNATIVE APPROACH**

- ASSUME PRIMARY LOAD PATH FAILS
- SET VISUAL INSPECTIONS BASED ON GROWTH FROM SMALLER FLAWS IN THE SECONDARY LOAD PATH.

**THIS APPROACH ALSO DOES NOT WORK**

- TEST SHOWS COMPLETE LIFT FRAME FAILURE OCCURS
- LOADS GO UP BY 2.5 TIMES IN THE SECONDARY LOAD PATH
- GIVES RF < 1.0 AT LIMIT LOAD
- RESIDUAL STRENGTH TYPICALLY 75% LIMIT LOAD

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**HOW COULD THE APPROACH BE MADE TO WORK?**

| CHANGE  | EFFECT   |
|---|--|
| REDUCE THE RESIDUAL STRENGTH REQUIREMENT TO LESS THAN LIMIT LOAD BASED ON LOW PROBABILITY OF OCCURRENCE | MIGHT MAKE SECONDARY LOAD PATH APPROACH VIABLE                                     |
| IMPROVE NDT CAPABILITY  | SMALLER INITIAL FLAWS BELOW THRESHOLD  |
| REAL-TIME MONITORING  | REDUCED INSPECTION COST BUT HIGH SYSTEM COST, COMPLEXITY, AND RISK OF FALSE ALARMS |
| ONLY CONSIDER LOCATIONS AT RISK OF DAMAGE   | REDUCED INSPECTION REQUIREMENT BUT MAY MISS SOMETHING                              |

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### THE EFFECT OF MANDATORY CRACK GROWTH

- Higher weight
- Designed from the start with higher redundancy
- Use of composite materials for primary structure

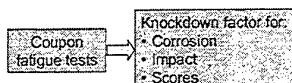
New aircraft qualified to these requirements will be penalised compared with older aircraft which may even lack a thorough Safe Life evaluation.

To improve safety the weight would be better used in the dynamic components or in HUMS systems.



### ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

"The capability of flawed structure as shown by tests or analysis based on tests to sustain, without measurable flaw growth, the spectrum of operating loads expected during the service life of the rotorcraft or during an established replacement time."



### ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

#### Initial Flaw Requirement

- Largest realistic flaw that would not be picked up by routine inspections.
- Ideally needs to be based on in-service experience
- Difficult for new aircraft with new materials and design concepts



### ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

#### Coupon Test Programme

| Damage         | Application Method  | Levels                                   | Number of Specimens |
|----------------|---|--|---------------------|
| Datum          | No damage   | -  | 10                  |
| Score          | Machining tool dragged across the specimen.   | 0.125mm deep<br>0.25mm deep              | 10<br>10            |
| Sharp Impact   | Pyramidal impact was applied perpendicular to the coupon  | 25J<br>12.5J                             | 11<br>5             |
| Oblique Impact | 10mm wide screwdriver impact at 45° to test section   | 6J                                       | 10                  |
| Corrosion      | Specimen placed in salt environment for 2 days. Damage site pre-drilled to represent a corrosion pit. | 0.4mm deep<br>0.2mm deep<br>No pre-drill | 10<br>10<br>3       |



### ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

#### Coupon Design

240 mm



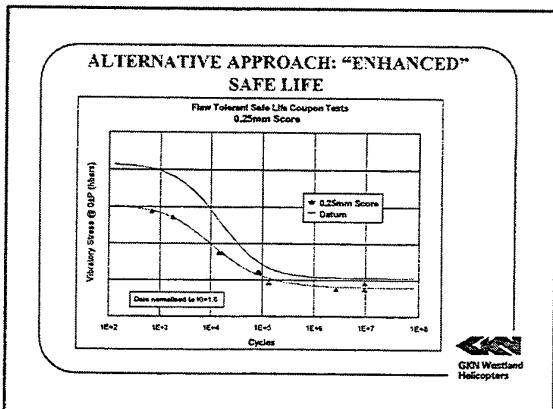
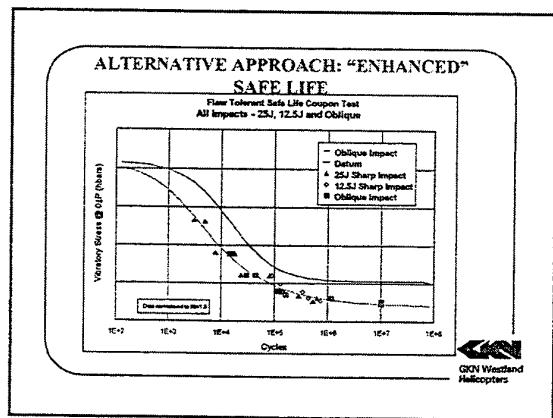
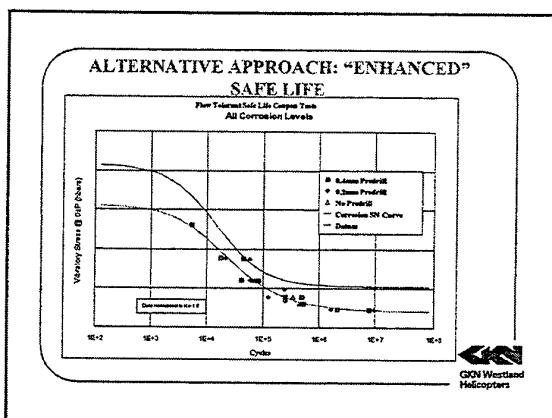
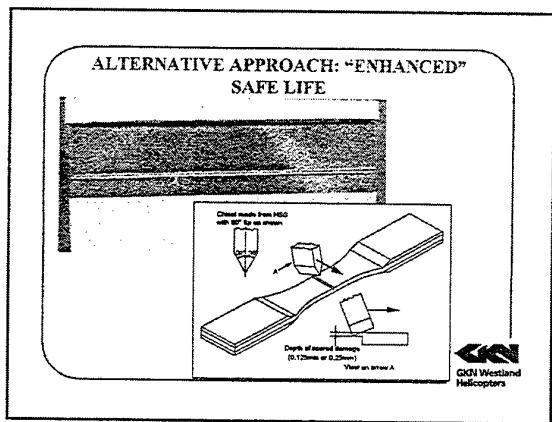
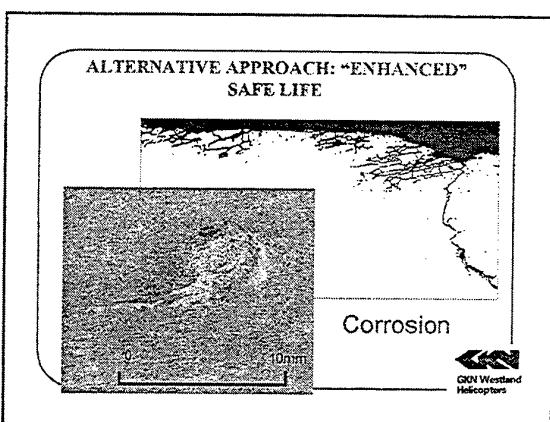
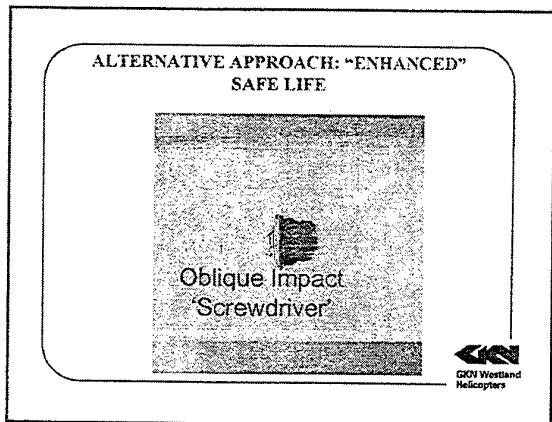
### ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

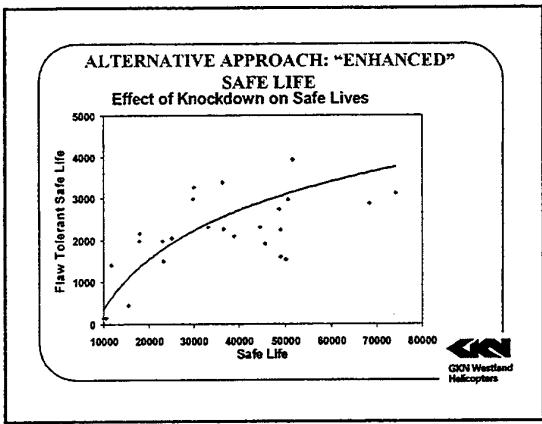
#### Sharp Impact

25 Joule

12.5 Joule







**CONCLUSIONS**

- This case study illustrates the difference between the Safe Life approach and crack-growth-based damage tolerance.
- GWHL research has shown that many crack growth models are inaccurate and non-conservative under helicopter loads.
- Near-threshold and short crack behaviour are essential elements of accurate predictions.

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**CONCLUSIONS - continued**

- Our comprehensive test and analysis programme has shown that crack-growth damage tolerance will not work on conventional rotorcraft structures even if they have good, rigorously demonstrated, Safe Lives.
- This is because of their design and high R-ratio (low threshold) + high number of cycles load spectra.
- Service experience shows that more onerous design requirements or inspection regimes are not warranted.

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**CONCLUSIONS - continued**

- The Enhanced Safe Life approach represents the best approach to Damage Tolerance for helicopters provided that the ESL is used as an inspection period.

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# **Crack Growth Rates under Vibratory Loading for Helicopter Damage Tolerance**

**Richard Buller**  
Senior Stress Engineer  
**ENGAGE**

Engage is a GKN company

*Cranfield*  
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## Overview

- Damage tolerance in helicopters under FAR 29.571.  
Differences with fixed wing methods.
- Vibratory loads - static and dynamic components.
- Helicopter component load spectrum development.
- Study of vibratory load crack growth using omission level technique.
- Discussion and conclusions.

## Helicopter damage tolerant design

Differences between application of fixed wing and helicopter crack growth technology:

- Highly variable loading experienced by components due to wide variety of manoeuvre loads and flight sorties.
- High accumulation of vibratory load cycles originating from main rotor revolution.
- High yield strength materials selection based on fatigue initiation properties.

The application of the fail-safe design considering flaw growth procedure to the damage tolerance analysis of helicopter components can not simply be translated from the well established methods of damage tolerance design used in fixed wing aircraft since the 1970's. For instance most helicopter structures have one main load path and so there is little opportunity of a redundant, fail-safe design. There are several important differences between the fixed wing and helicopters which impact on process of design against fatigue.

## Crack growth under vibratory loading

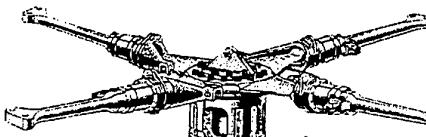
- Source of vibratory loads mostly from main rotor at  $1R$  for rotor components and  $nR$  (blade passing frequency) for fuselage.
- Vibratory loads are superimposed on manoeuvre loads so are typically high mean stress ( $R<0.7$ ).
- Vibratory loads can be removed for testing or analysis.
- Experimental omission level technique developed to study crack growth mechanisms under vibratory loads.

The vibratory cycles typically make-up 90% of the fatigue loads in a helicopter loading spectrum. Therefore, knowledge of the behaviour of these cycles is essential in providing an accurate means to determine inspection periods and to ensure that the helicopter service life is achieved without premature failure due to fatigue.

The understanding of the effect of vibratory load cycles on the crack growth rates in helicopter components under spectrum loading is two-fold: (1) Large reductions in laboratory and full-scale fatigue testing time can be achieved by omitting low amplitude, vibratory cycles below a certain stress cycle range from the loading spectrum using the threshold stress intensity property. (2) The point at which these cycles start to cause significant fatigue crack damage indicate the ability to assign reasonable inspection periods for a component.

However, to do these correctly the crack growth rates of the vibratory load cycles need to be established through consideration of load interaction effects, crack lengths and applied stress levels.

## Rotorhead stress sequence development



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Lynx main rotor



- Stress sequence for Lynx main rotorhead based on methodology of Felix/Helix. Strain gauge data taken from flight test data.
- Represents 140 sorties and 190.5 flying hours.
- Contains 1,900,000 1R vibratory load cycles.

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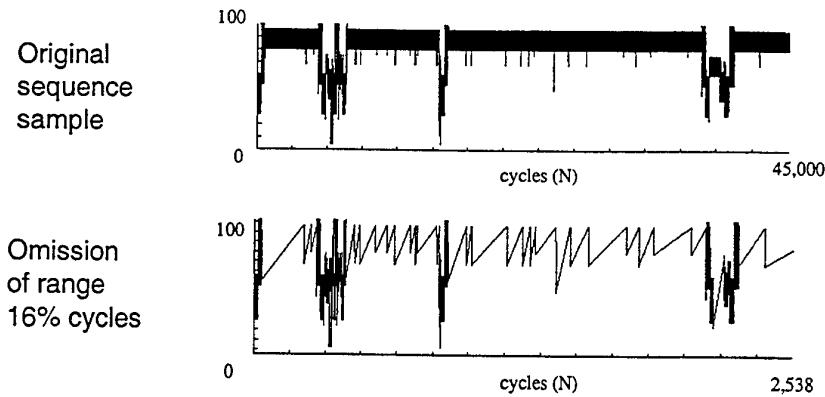


The representative stress sequence used for the testing of omission level effects was developed for a helicopter rotorhead component using the well established methods of Helix and Felix. These sequences consist of 140 sorties representing 190.5 hours of flight. Each sortie in the sequence represents either training, transport, anti-submarine warfare (ASW) or search and rescue (SAR) flight. These sorties are in turn described by a defined sequence of twenty two manoeuvre types and are given in three flight lengths of 0.75, 2.25 and 3.75 hours.

The Felix stress sequence is based on a fixed rotor helicopter so was chosen to be the basis for the development of a stress sequence at a location on the fixed main rotor component studied here. The critical location on the rotorhead component was strain gauged and data were recorded during several test flights to provide data for the stress sequence generation. The strain gauge data were then processed and assembled in accordance with the procedures used in Felix. By using the Felix procedure the small, IR vibratory loads which the component experiences due to rotor rotation are well defined and can be used for testing on laboratory specimens.

One drawback of the Felix procedure is that the load data are strictly defined into integer levels of four between 0 and 100 so that the sequence not as 'continuous' as desired.

## Progressive omission level technique



- Retains sequencing of major load cycles.

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A novel method was devised to determine the fatigue damage contribution of the different cycle ranges by progressively omitting small range cycles of increasing range from the rotorhead loading sequence. The method considers four omission levels at 16, 20, 24 and 32% cycle ranges. The 'peak-valley filtering' method was used to omit the cycles at each step where trough-peak pairs are retained or omitted according to the desired load range. It maintains the correct ordering of the sequence as is shown in Figure.

The method assumes that the difference in crack growth rates between the two omission level tests will be due to the fatigue crack damage of the omitted cycles. Therefore, the crack growth damage rates of various cycle ranges can be separated out and used for analysis.

## Summary of testing

- Four omission level tests each on titanium Ti-10V-2Fe-3Al and aluminium 7010 T73651 compact tension specimens (17.5mm).

| Omission range level | Rainflow cycle count | Reduction in length from original (%) |
|----------------------|----------------------|---------------------------------------|
| 16%                  | 1989925              | -                                     |
| 20%                  | 113063               | 94.3                                  |
| 24%                  | 110907               | 94.4                                  |
| 32%                  | 51404                | 97.4                                  |

- Crack length monitored with automated direct current electro-potential method.
- Instron digitally controlled machine for spectrum loading.

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### Specimens

Ti1023 was used because this is the material from which typical main rotorhead components are manufactured from. The lower strength 7010 was used to compare the effect of yield strength differences on load interaction effects between the two alloys. 7010 is an alloy typically used in the manufacture of helicopter fuselage structures. The CT specimens were  $t=17.5$  mm thick giving predominantly plane strain stress states throughout all the tests.

### Crack Length Measurement

A custom designed direct current potential drop (DCPD) crack length measurement system was designed to provide measurement of fatigue crack growth rates for all the fatigue tests.

### Loading

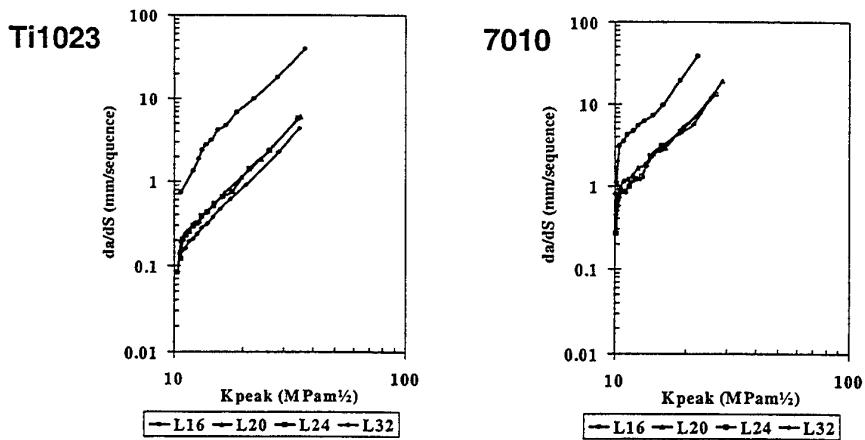
Loading was performed under load control, at 10Hz and under ambient laboratory conditions. The tests were carried out in accordance with the procedures defined in ASTM E 647-93 "Standard Test Method for Measurement of Fatigue Crack Growth Rates".

Four tests on each of the Ti-1023 and 7010 CT specimens were conducted at each of the omission levels defined in the table.

Loads are defined in terms of the peak or maximum load experienced by the sequence which is level 100 in the Felix definition. Hence the peak stress and stress intensity factor are referenced to this level and are termed  $S_{peak}$  and  $K_{peak}$  respectively

## Omission level crack growth rates

- Removal of vibratory cycles (16%) gives increase in crack growth lives of 3.6 for 7010 and 5.4 for Ti1023.



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For both materials the effect of progressively omitting small range cycles is to increase the flights to failure for the specimens. Removal of the level 16 cycles from the full sequence test gave an increase in life of 3.6 and 5.2 times for 7010 and Ti1023 respectively. In each material the number of flights to failure were almost identical for the 20% and 24% omission tests as expected because there are a relatively small number of 24% cycles (2156) and these caused little fatigue damage.

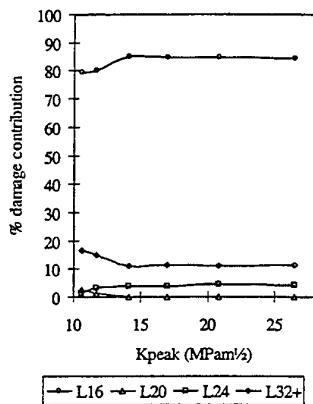
The increase in specimen life due to omission of small cycles has been observed by others and is due to a removal of damaging load cycles. In this case the dominant omission level effect is due to removal of the vibratory load cycles which is in part due to the large proportion of small cycles in the sequence.

Figures show the crack growth rates per sequence for each material against the peak stress intensity factor,  $K_{peak}$ . The effect of omitting the 16% vibratory cycles is clearly shown for each material as the growth per sequence is reduced significantly for all the 20%, 24% and 32% sequences indicating the progressive omission of cycles from each were causing fatigue crack damage in the previous test.

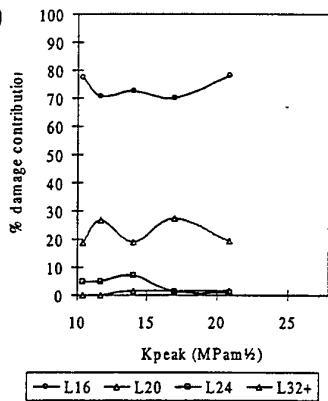
## Crack growth damage contributions

- Difference in crack growth rates between omission level test due to crack length damage of omitted vibratory cycles.
- Correlated with SEM fractographic examination of fracture surfaces.

Ti1023



7010



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It may seem obvious that the removal of 94% of the cycles from the sequence would result in an increase in life to failure. However, most of the vibratory cycles operate in the near-threshold regime so would have near-zero growth rates and may not always contribute significantly to fatigue crack damage in a strictly linear manner due to the slope of the  $\log da/dN$  versus  $\log \Delta K$  curve.

The average crack growth rate per loading sequence is calculated at different levels of  $K_{\text{peak}}$  and from these the crack growth increment per omission cycle range can be calculated: The crack length damage contribution for each cycle range during the full sequence test (all cycles included) is the sum of all the individual cycle ranges based on the assumption that each cycle range causes the same amount of crack growth increment in each omission level test. That is, the differences between the omission level tests are due to the omitted cycles.

In both materials the 16% range cycles cause a large proportion of the damage over the entire test crack lengths. On average, the damage for the 16% cycles is about 85% for Ti1023 and 75% for 7010. The next largest contributor to crack growth damage are the 32% range and above cycles. The damage contributions are typically 15% for Ti-1023 and 20% for 7010. In both materials the 20% cycles give negligible contributions to crack damage which is expected because they only constitute 0.1% of the loading sequence

## Crack growth damage due to vibratory loads

- Modelling of omission level tests using a crack growth model without load interaction effects (ESACRACK/NASGRO) gives non-conservative results.

| Material | Omission | $a_i$ | $a_f$ | Test $N_T$ | Model $N_M$ | $N_f/N_M$ |
|----------|----------|-------|-------|------------|-------------|-----------|
| Ti1023   | 16       | 17.43 | 49.8  | 1145       | 3629        | 0.32      |
|          | 20       | 17.43 | 49.2  | 6341       | 6107        | 1.04      |
|          | 24       | 17.43 | 49.5  | 6440       | 6129        | 1.05      |
|          | 32       | 17.43 | 48.6  | 8612       | 8496        | 1.01      |

- Vibratory load cycles grew faster than expected under sequence loading conditions when compared with constant amplitude loading data.

It is significant that, even though the 16% range cycles constitute 94% of the total number of cycles in the sequence, the contribution to crack growth damage by the vibratory cycles is higher than that which a typical model would predict.

The 16% cycles had *greater* fatigue crack growth rates under helicopter sequence test loading than under equivalent CAL conditions. This is an important observation because most fatigue crack models would not predict that these smaller cycles grow faster under CVAL as they are usually 'tuned' for retardation. Hence prediction of growth rates under helicopter CVAL could give misleading predictions.

The Esacrack analysis (Table) indicates the non-conservatism of the modelling of the Ti-1023 full sequence test by a factor of three.

## Consequences for helicopter damage tolerance

Accelerated growth rates of high R ratio vibratory load cycles will impact on helicopter damage tolerance design for the following:

- Design of fatigue loading sequences for analysis and testing.
- Selection of materials and component stress levels.

Simplification of a helicopter loading sequence for testing and analysis is best achieved by omission of the small vibratory load cycles which gives a significant reduction in sequence lengths. However, the main drawback demonstrated here is that these small cycles contribute to a significant amount of crack growth damage so modelling can be misleading. A criteria for 'safe' omission of small load cycles could be based upon the mechanism of accelerated crack growth of these cycles. This would be done by determining when a cycle range can be omitted from a sequence such that its omission will not result in changes in specimen life - best performed by testing.

The damage tolerance capability of a helicopter structure can be demonstrated by examining crack growth rates over a range of flaw sizes for an arbitrary edge cracked component under realistic service stresses.

## Omission techniques for helicopter spectra

- Observations indicate that vibratory load cycles have accelerated growth rates under spectrum loading.
- Increased  $\Delta K_{\text{eff}}$  ( $K_{\max}-K_{\text{cl}}$  or  $K_{\max}-K_{\text{PR}}$ ) compared to constant amplitude loading.
- Due to frequent underloads in sequence (GAG and manoeuvre) and absence of tensile overloads.

These observations indicate that the small range cycles have accelerated growth rates under CVAL compared with the equivalent cycles under CAL. An explanation of this observation can be based on transient residual stress fields ahead of the crack tip give an increased  $\Delta K_{\text{eff}}$  when compared with constant amplitude loading is proposed.

The key to the accelerated growth rates is the 'form' of the loading sequence. Typically most cycles have similar  $K_{\max}$  values, there are no tensile overloads and periodic underloading occurs due to the GAG cycles and low R ratios manoeuvres.

Underload tests have shown that the underload can eliminate or reduce the effect of overloads during high R ratio loading and can cause acceleration during constant amplitude loading. The effect of intermittent underloads causing acceleration of high R ratio CAL crack growth rates has been observed by others. However, the GAG underloads occur infrequently so it is most likely that underloading due to the occasional low R ratio manoeuvre is responsible for increasing the  $\Delta K_{\text{eff}}$  of the high R ratio, vibratory cycles.

## Omission techniques for helicopter spectra ...

- 'Safe' omission criteria based on:

$$(K_{\max} - K_{PR}) < K_{th}$$

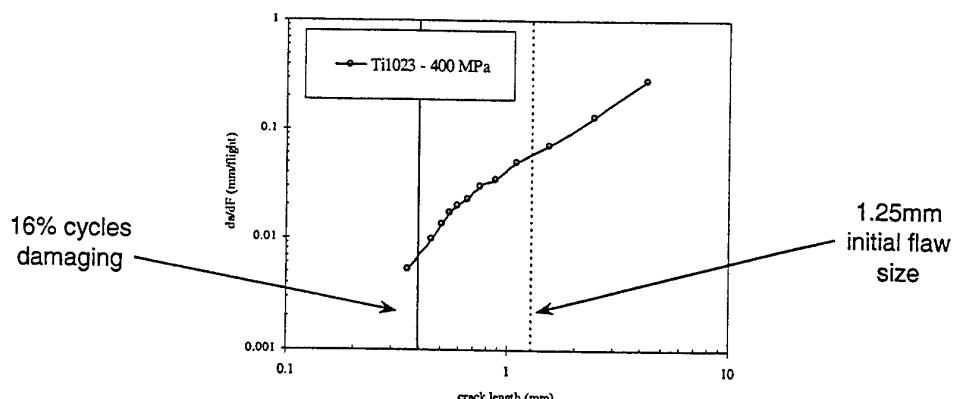
considering cycle range, mean and threshold.

- Careful determination and modelling of near-threshold region over variety of R ratios and considering load-interaction effects.
- 'Safe' omission for crack growth much less than for initiation based methods

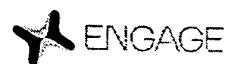
The vibratory loads and the effect of these on fatigue crack growth using the fatigue loading sequence provides a useful means of understanding the mechanisms under which cracks grow in helicopter components. As the concurrent conditions of plane strain, near threshold and high R ratio affecting crack growth are unique to helicopters the mechanisms of growth under such conditions can be determined by analysing the crack growth behaviour of vibratory loads through omission. By being able to accurately predict the behaviour of vibratory load cycles, which dominate crack growth damage, a model with general applicability to helicopter damage tolerance design can be developed.

## Material and component stress levels

- Damage tolerance capability demonstrated by considering spectrum crack growth rates for arbitrary Ti1023 rotor component,  $S_{peak}=400\text{ MPa}$ .



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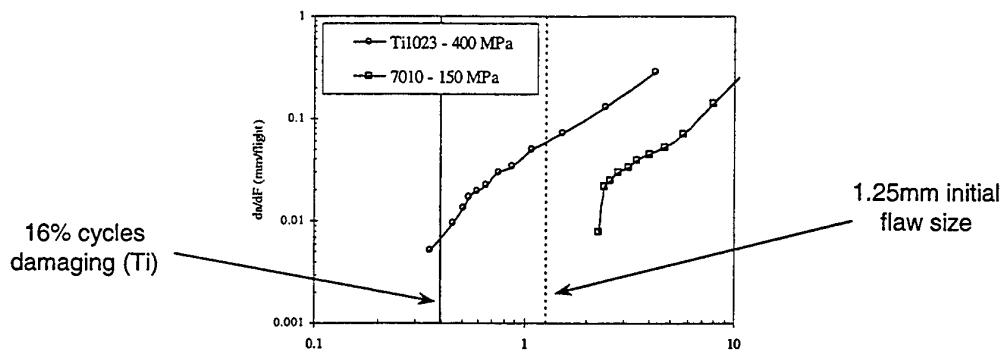


This example component is 'subjected' to the full test sequence with an operating peak tensile stress of 400 MPa (approximately one third of the  $\sigma_{UTS}$  for Ti1023) by converting the crack growth rate per sequence curve into crack lengths that will produce the same  $K_{peak}$ 's assuming similitude exists. The experimental data can be applied to range of crack lengths from 0.35 to 4 mm.

Figure indicates the crack length above which the vibratory load cycles will start to cause damage. At 0.39 mm this is well below the typical initial starting defect size of 1.25mm used in the aerospace industry. Improving the damage tolerance capability of a helicopter structure would require the vibratory load cycles to be maintained below their fatigue threshold. Elimination of crack growth due to vibratory load cycles would require a shift of the curve to the left and/or reducing the size of the initial defect. This could also mean a reduction in component peak stress level which usually translates into an increase in weight

## Material and component stress levels...

- Comparison between 'typical' rotorhead (Ti1023) and fuselage (7010) components with  $S_{peak}=150\text{ MPa}$  for 7010.



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Design strength and crack growth rate differences between 7010 and Ti1023 are also indicated in Figure. A typical maximum operating stress of 150 MPa was used for 7010.

This emphasises the differences between different parts of the helicopter structure because some materials will be more sensitive to crack growth under vibratory load cycles. This is mainly because of the lower operating stress that a safe life design gives for aluminium alloys and not necessarily due to material fatigue crack growth properties. Note that a fuselage component would be subjected to the 4R vibratory loads.

Titanium or other high strength metals may have better fatigue crack growth properties but are subject to higher operating stresses. This indicates that aluminium fuselage structures would be amenable to damage tolerance design because vibratory load cycles will not grow from typical initial flaw sizes (1.25 mm).

A reasonable inspection regime could be applied because crack growth will occur under manoeuvre load cycles in a similar manner to fixed wing transport aircraft.

## Summary

- Progressive omission level technique used to investigate vibratory load cycle crack growth damage.
- Vibratory load cycles typically 94% of spectrum loads and contribute to 80% of the crack growth damage.
- Standard constant amplitude loading linear summation models are non-conservative by factor of three. ie. Accelerated growth under vibratory loading.
- Damage tolerant analysis of high strength rotor (dynamic) components difficult to achieve with current safe life design stresses.
- Damage tolerant analysis of aluminium fuselage (static) components can be achieved with current safe life design stresses.

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Small vibratory load cycles had the most significant effect on the overall growth rate of the variable amplitude loading sequence, causing up to 80% of the fatigue crack damage which was supported by fractographic examination. Comparison with a constant amplitude loading model revealed that these small cycle were growing at an accelerated rate when applied in conjunction with periodic underloads due to low R ratio underloading.

Significant crack growth rates due to vibratory load cycles under helicopter loading conditions mean that the damage tolerant design of high strength (Ti1023) helicopter components will not be achieved using current stress levels given by a safe life analysis. A reduction in stress levels and initial flaw size is required but with increases in component weight and inspection length capability.

Lower strength aluminium alloy (7010) components are less sensitive to vibratory load cycles because of reduced operating stresses given by a safe life analysis. This indicates that aluminium fuselage structures would be amenable to damage tolerant design because vibratory load cycles will not grow from typical initial flaw sizes (1.25 mm). A reasonable inspection regime could be applied.

## SESSION E

# CRACK GROWTH PREDICTION METHODS FOR HELICOPTER LOAD SPECTRA AND MATERIALS



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**DERA**



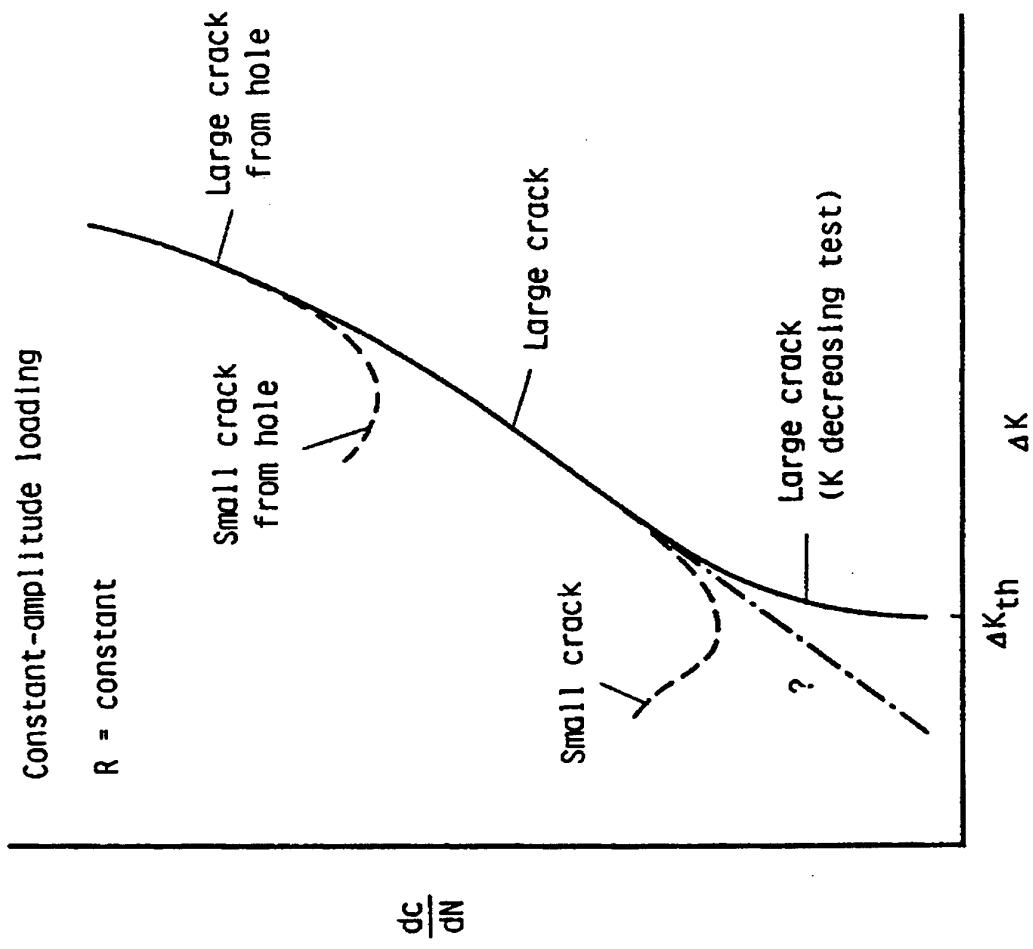
# THE LARGE-CRACK ANOMALY IN THE THRESHOLD REGIME

J. C. Newman, Jr.

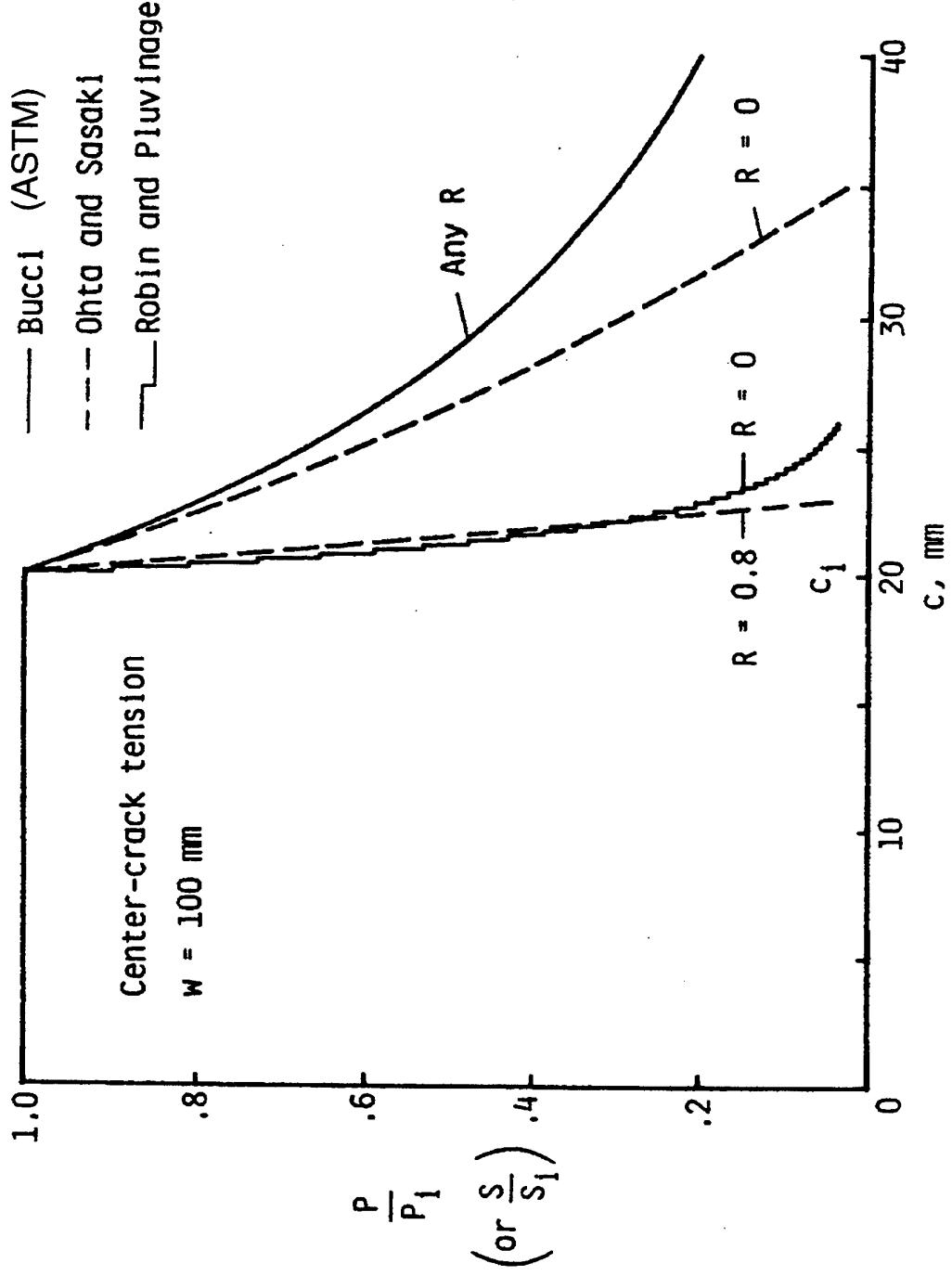
NASA Langley Research Center  
Hampton, Virginia USA

Workshop on Damage Tolerance in Helicopters  
Cranfield University  
Cranfield, United Kingdom  
April 4-5, 2000

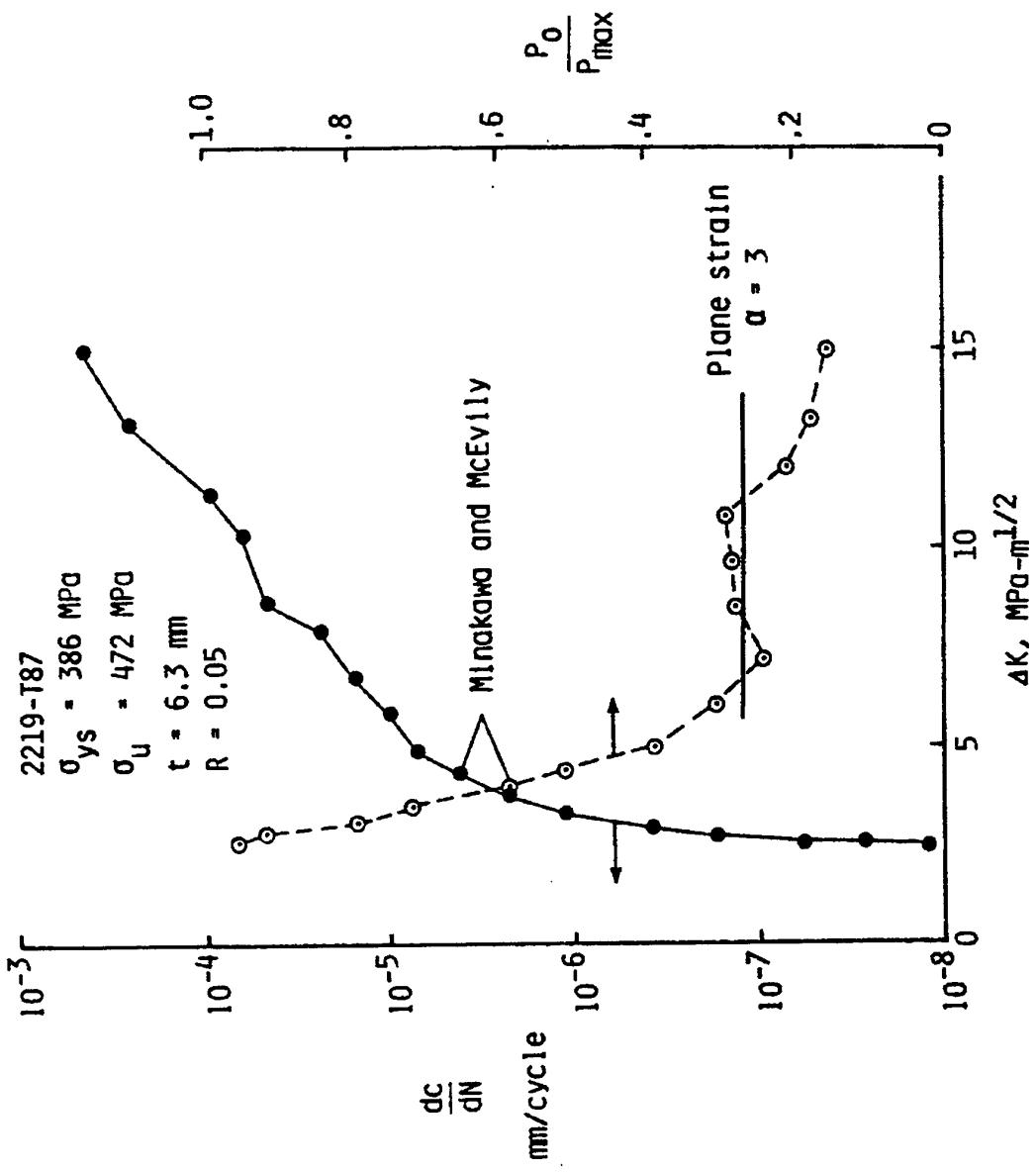
# TYPICAL LARGE- AND SMALL-CRACK DATA



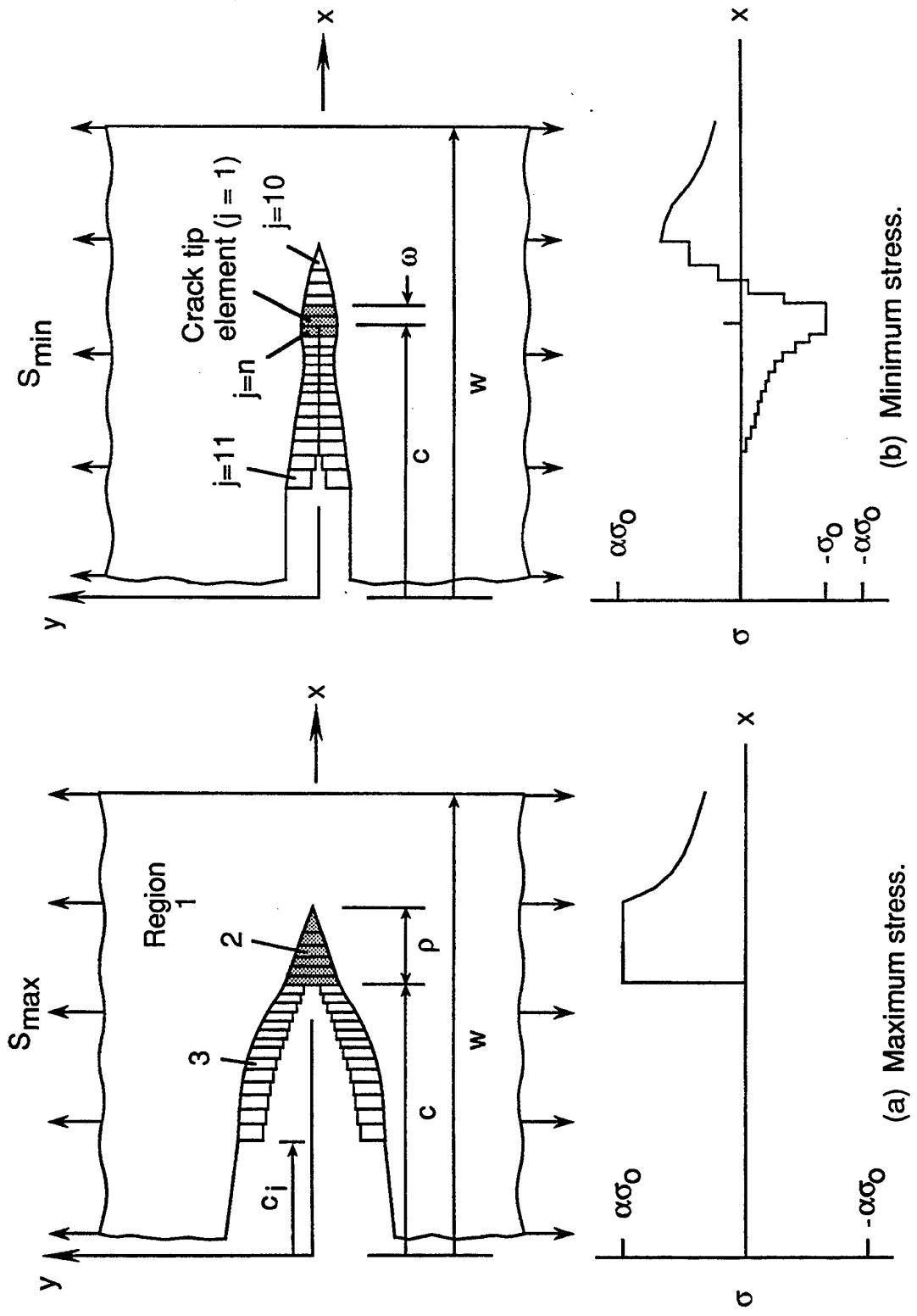
# TYPIICAL LOAD-REDUCTION PROCEDURES



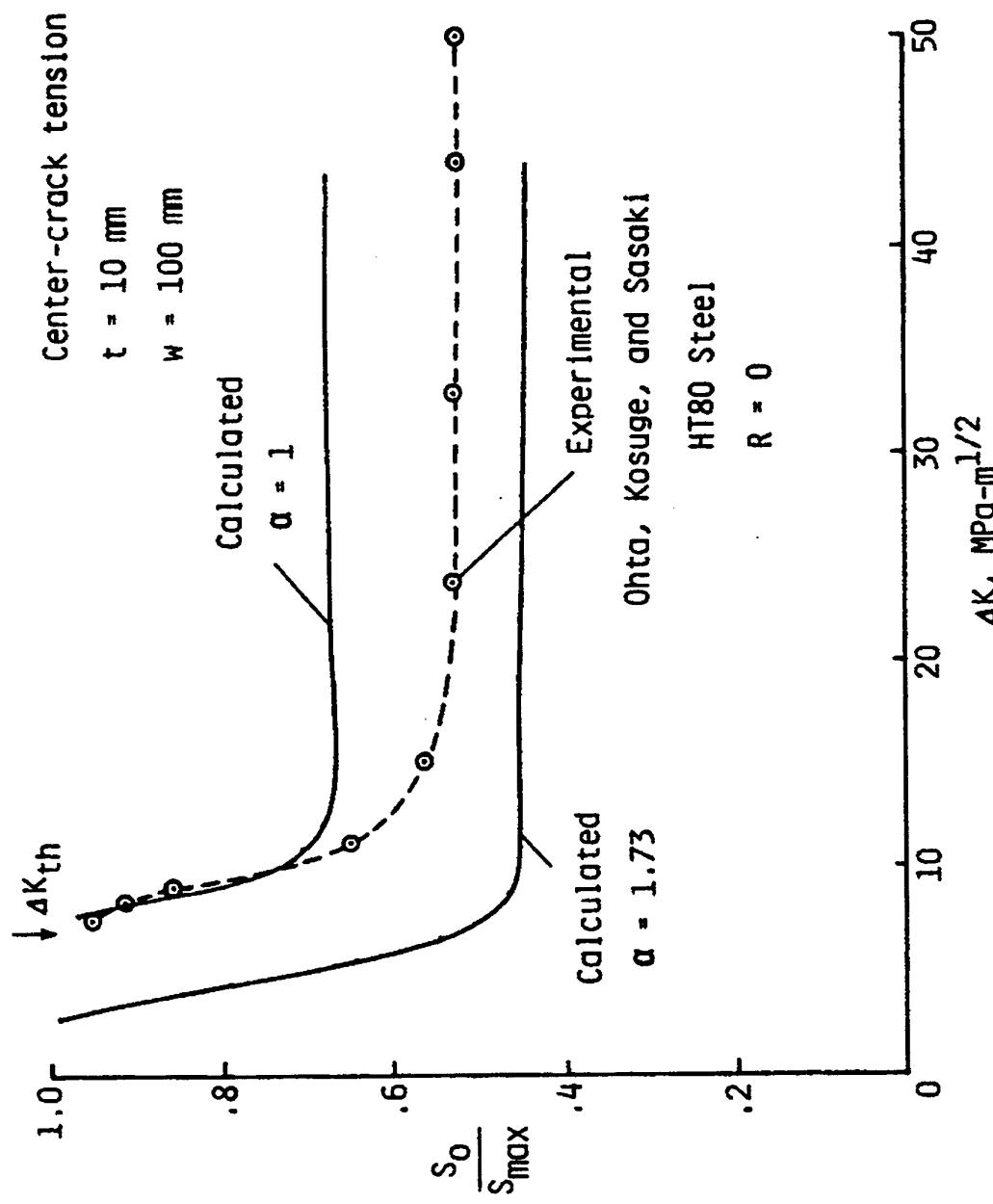
# MEASURED CRACK-GROWTH RATES AND CRACK OPENING LOAD LEVELS



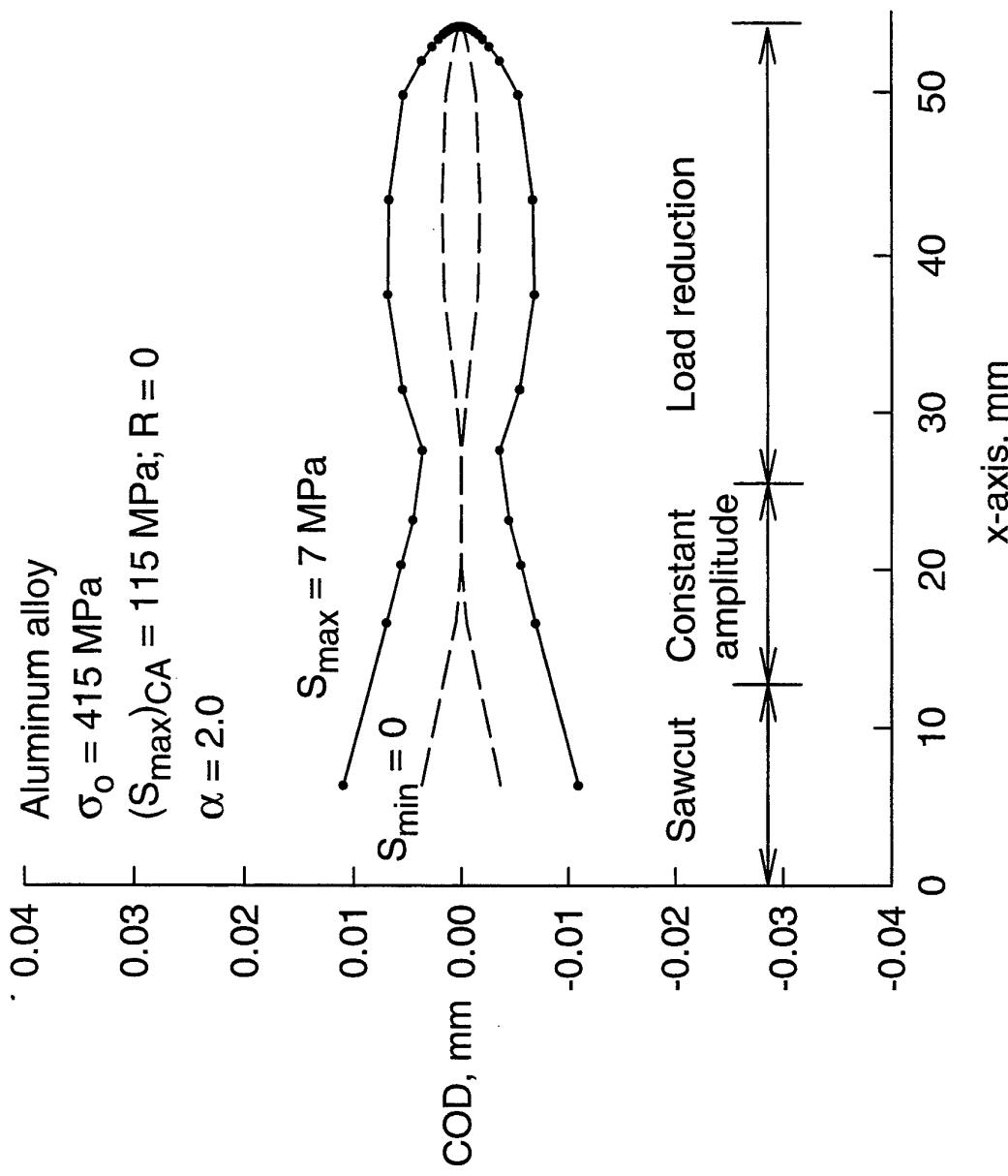
# FASTTRAN -- CRACK CLOSURE MODEL



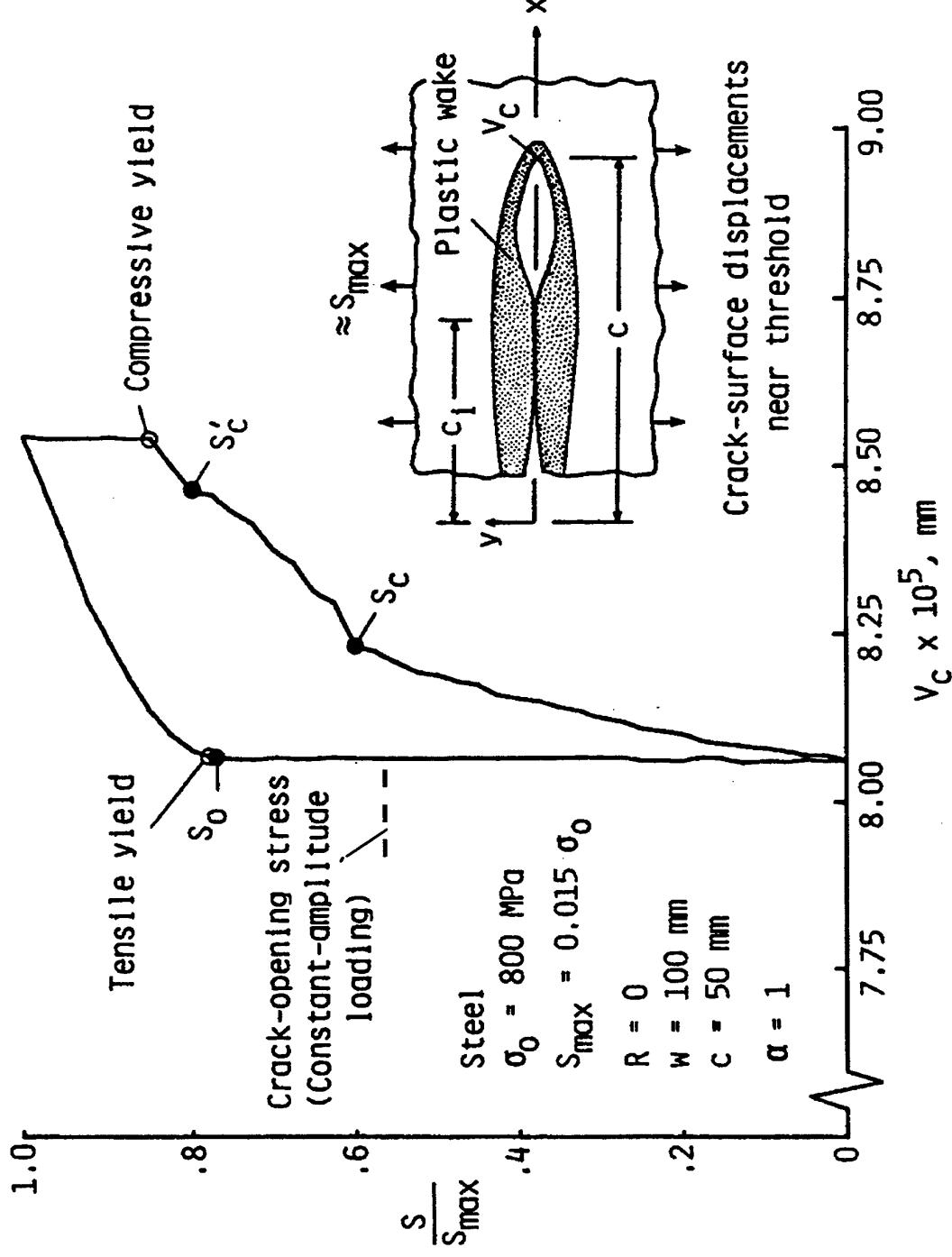
# COMPARISON OF MEASURED AND CALCULATED CRACK OPENING LEVELS DURING LOAD REDUCTION



# CRACK-SURFACE DISPLACEMENTS AFTER LOAD-REDUCTION TEST SIMULATION

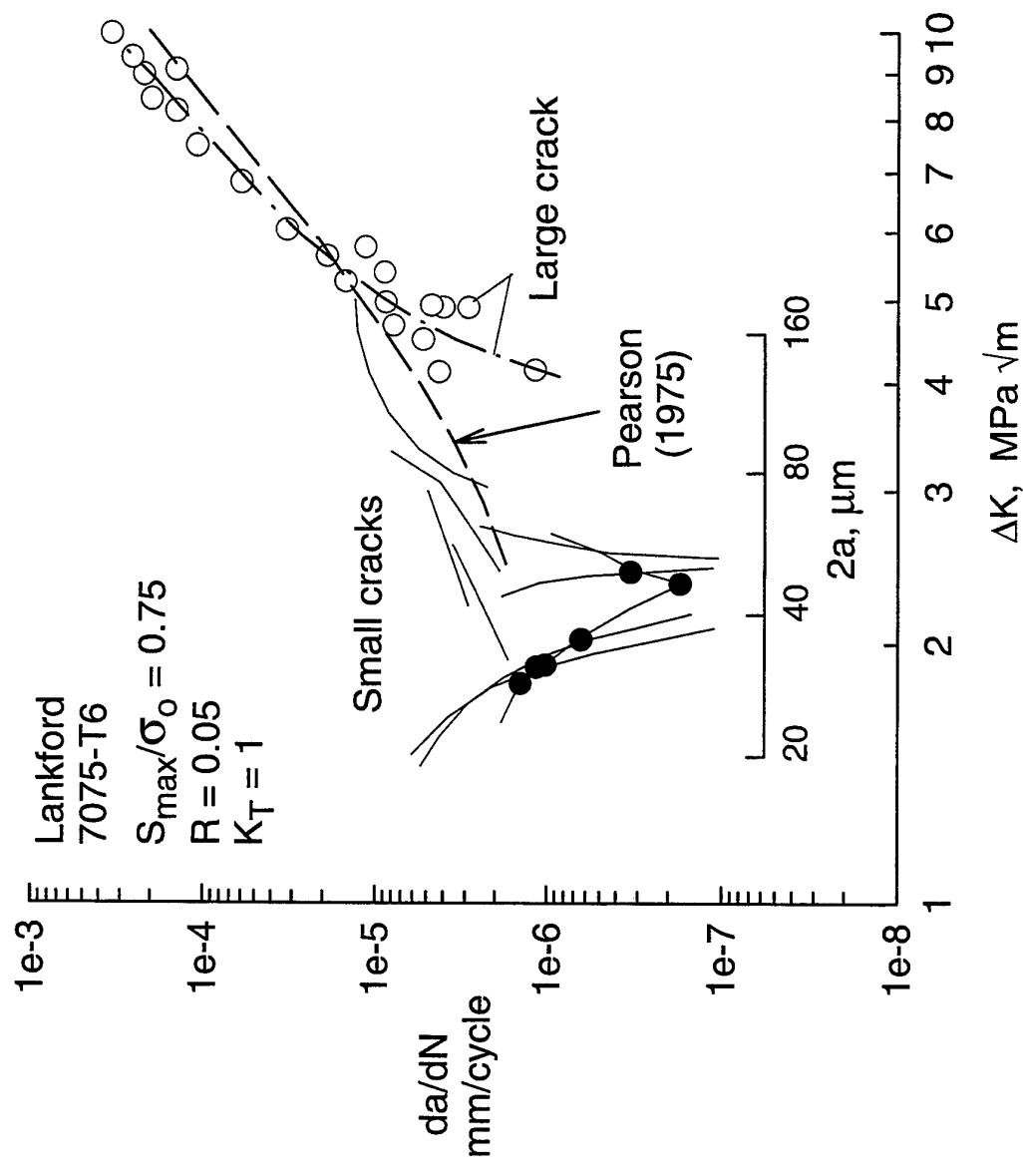


# CALCULATED CRACK-TIP DEFORMATIONS AFTER LOAD-REDUCTION PROCEDURE

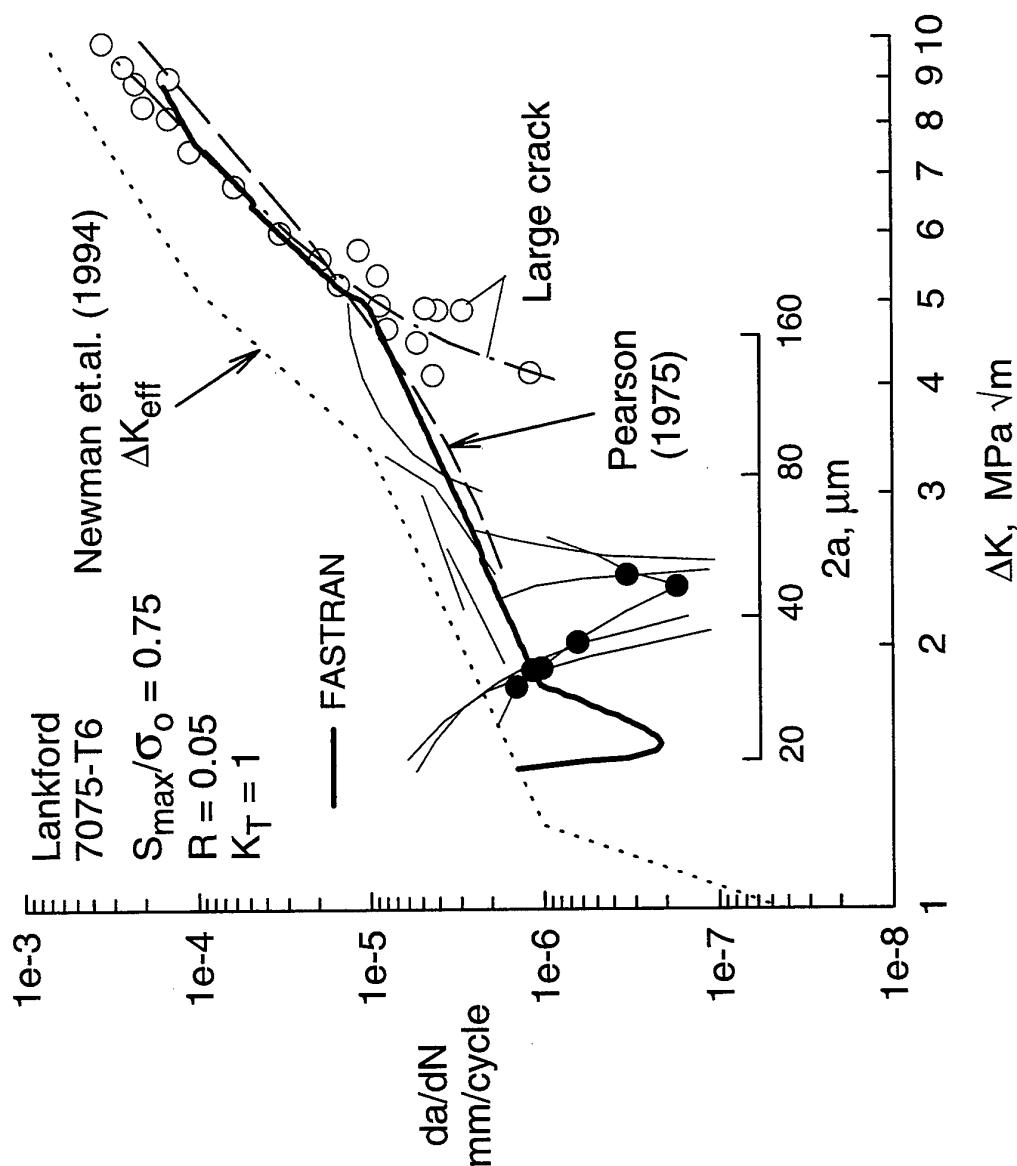


# LARGE- AND SMALL-CRACK DATA ON 7075-T6

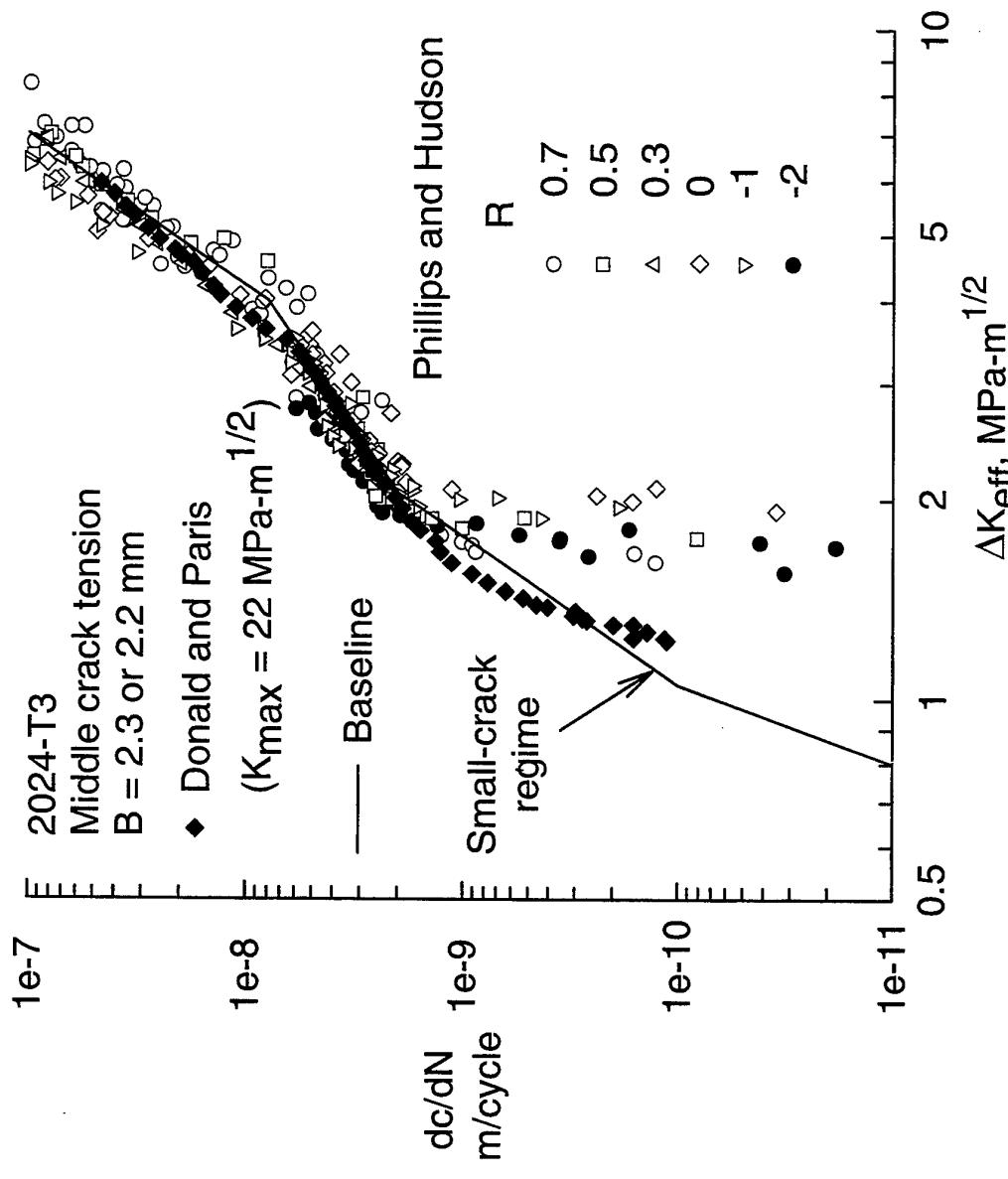
Pearson 1975; Lankford 1983



# COMPARISON OF LARGE- AND SMALL-CRACK DATA AND CRACK-CLOSURE MODEL PREDICTION



# COMPARISON OF LOAD-REDUCTION, $K_{\max}$ -EQUAL-CONSTANT, AND SMALL-CRACK RESULTS



# CONCLUDING REMARKS

- Load-reduction test procedures on large cracks cause “remote” closure that elevates the elastic stress-intensity factor range threshold ( $\Delta K_{th}$ ).
- Small-crack effects (lack of closure) occur in the micro-crack length range (< 100  $\mu\text{m}$  or 0.004 inches) for a wide variety of materials.
- Small-crack data for cracks greater than about 100  $\mu\text{m}$  (0.004 inches) in length is “steady-state” crack-growth behavior for a wide variety of materials.
- Small-crack data is “real” and large-crack data in the threshold regime is the “anomaly”.

*Workshop on Damage Tolerance in Helicopters  
Cranfield, UK, 4-5 April 2000.*

## THE BASICS OF A MODEL FOR FATIGUE CRACK GROWTH IN HELICOPTER COMPONENTS

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Helicopters have traditionally been designed using the “safe life” design philosophy. For about 10 years, the “damage tolerant” design philosophy, successfully applied to fixed wing aircraft design and maintenance procedures [1,2], is either discussed or applied to rotorcraft structures [3,4]. A flaw tolerant design philosophy requires the calculation of crack growth due to service loading with sufficient accuracy. Moreover, the crack growth phase has to be long enough to realise an inspection procedure on a cost effective basis. The main requirements to a crack growth model are robustness (simplicity, parameters are easy to obtain, etc.) and a most accurate mirroring of the physics of crack growth. A fair amount of research has been done in the field of fatigue crack growth and a number of crack growth prediction models exist which are based on different theoretical considerations. Despite the variety of crack growth models, a unified approach to fatigue crack growth is not yet found. The existing models involve various fitting parameters that adjust the result of the calculation to experimental crack growth data from the sequences they should predict. As long as no reliable crack growth code exists, a model that uses some “fitting parameters” is always superior to a complicated model in solving practical problems on a daily basis. It is no surprise that, e.g., the models by Wheeler [5] and Willenborg [6] are so widely established and used. Acknowledging that prediction models involving fitting factors have their place in quickly solving practical problems, we should try to understand the physics of fatigue crack growth. A physically correct model has to predict fatigue crack growth without extensive fitting, thus providing the basis for safer damage tolerant design and management of engineering structures.

This presentation tries to contribute in this effort and puts a new crack growth prediction model for metals into the discussion. Experimental evidence obtained on the aluminium alloy Al 7475-T7351 are generalised to result in a new fatigue crack growth prediction model for variable amplitude loading. The difference to other models is that the central parameter of this model is the crack propagation stress intensity factor,  $K_{PR}$ . This is the stress intensity factor at which the crack starts to propagate during the loading part of a cycle. The general description of the relatively simple loading sequences in terms of  $K_{PR}$  leads to a scheme that is denoted as the “Fatigue Crack Growth Map.” This map defines all possible crack growth conditions and is the heart of the crack growth prediction method. The Fatigue Crack Growth Map is generated for different alloys and temperatures. It is shown that any arbitrary loading spectrum consists only of three different types of load cycles. The model determines  $K_{PR}$  cycle by cycle throughout a

loading spectrum, which determines the driving force,  $\Delta K_{eff}$ , for the following loading cycle. The model requires the knowledge of the yield strength and the intrinsic threshold value,  $\Delta K_T$ , for the material, as well as five different functions that also depend on the material. Phenomenologically, the model accounts for residual compressive stresses ahead of the crack front, the *intrinsic* response of the material, crack closure, which is an *extrinsic* effect, and the *intrinsic* threshold value  $\Delta K_T$ . A few variable amplitude loading sequences are presented to compare predictions of  $K_{PR}$  with experiments. The results are in encouraging agreement.

In the following the viewgraphs of this presentation are listed. The interested reader will find a more detailed description of the model in [7,8]. Also References [9] and [10] will be needed, as well as [11] for more evidence on the Fatigue Crack Growth Map.

1. *Proceedings of the first joint DoD/FAA/NASA Conference on Aging Aircraft*. 1997, Ogden, Utah.
2. Various Authors, Aging of U.S. Air Force Aircraft. *Final Report of the Committee on Aging U.S. Air Force Aircraft, National Materials Advisory Board, Commission on Engineering and Technical Systems, National Research Council*, Publication NMAB-488-2, National Academic Press, Washington, D.C., USA., 1997.
3. An Assessment of Fatigue Damage and Crack Growth Prediction Techniques, *AGARD Report 797*, 1994.
4. Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft, *RTO-MP-24*, Feb. 2000.
5. O.E. Wheeler, Spectrum loading and crack growth. *Journal of Basic Engineering*, American Society for Mechanical Engineering Transactions, Vol. 94, 1972, 181-186.
6. J. Willenborg, R.M. Engle, H.A. Wood (1971) A crack growth retardation model using an effective stress concept. *AFFDL-TM-71-1-FBR*, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, USA.
7. M. Lang, A Model for Fatigue Crack Growth, Part I: Phenomenology, Accepted for Publication in *Fatigue and Fracture of Engineering Materials & Structures*, appears in 2000.
8. M. Lang, A Model for Fatigue Crack Growth, Part II: Modeling, Accepted for Publication in *Fatigue and Fracture of Engineering Materials & Structures*, appears in 2000.
9. M. Lang, Description of load interaction effects by the  $\Delta K_{eff}$ -concept. *Advances in Fatigue Crack Closure Measurement and Analysis, ASTM STP 1343*, American Society for Testing and Materials, 207-223, 1999.
10. M. Lang and X. Huang, The influence of compressive loads on fatigue crack propagation. *Fatigue and Fracture of Engineering Materials & Structures*, Vol. 21, No. 1, 1998, 65-83.
11. M. Lang, J.M. Larsen, Fatigue crack propagation and load interaction effects in a titanium alloy. *Fatigue and Fracture: 30<sup>th</sup> Volume, ASTM STP 1360*, American Society for Testing and Materials, 201-213, 2000.

# The Basics of a Model for Fatigue Crack Growth in Helicopter Components

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*Workshop on Damage Tolerance in Helicopters, Cranfield University,  
Cranfield, Bedfordshire MK 430AL, England, on 4-5 April 2000.*



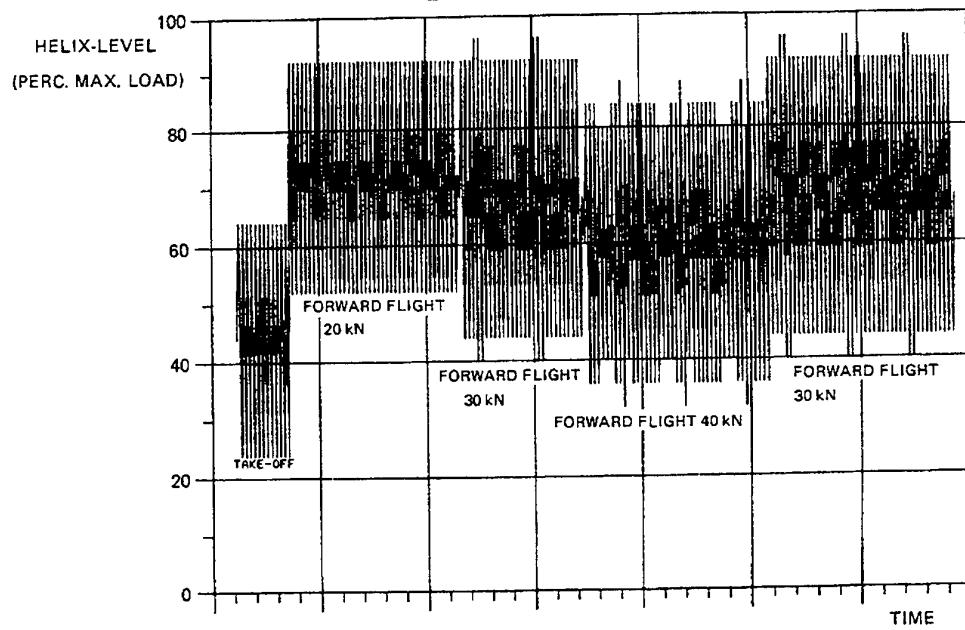
## *Damage Tolerant Design & Maintenance*

- Critical Location
- Crack Shape, K-Solutions
- Size of Initial Failure and Distribution
- Loading Spectrum
- **Fatigue Crack Growth Prediction**
- Confidence
- Corrosion, Temperature, Humidity
- Maintenance / Inspection Procedure (NDT)

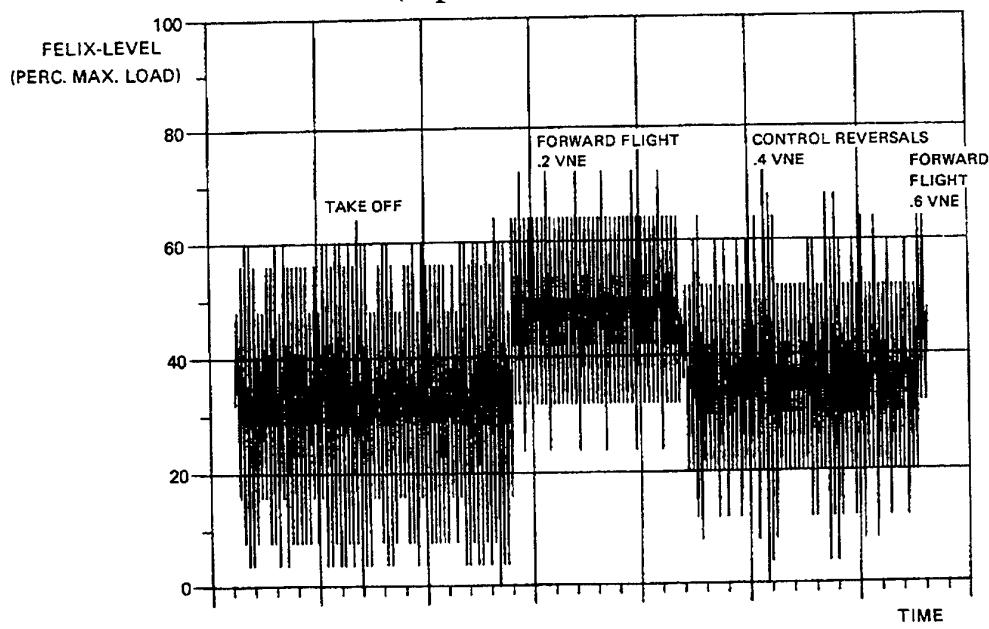
## *Inspection Procedures (Intervals and NDE Tools)*

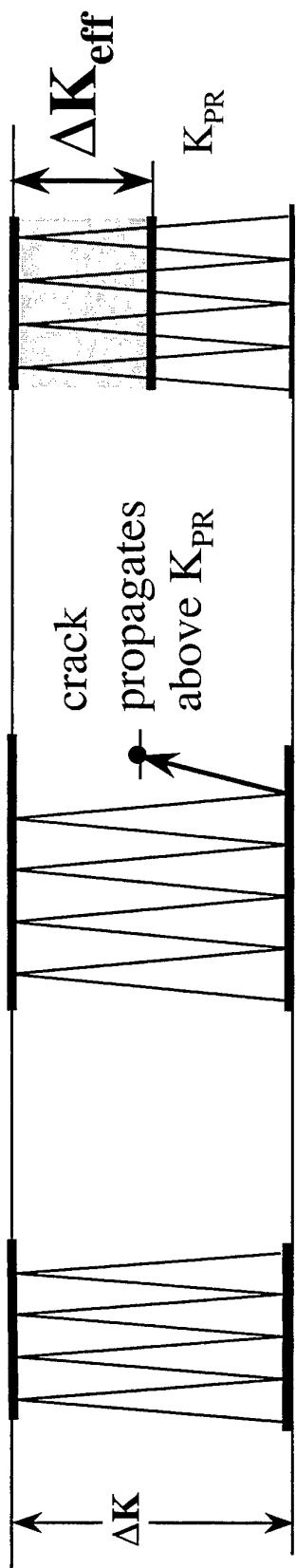
*Probability of Failure,*

**HELIX (represents first 90 sec. Training)**

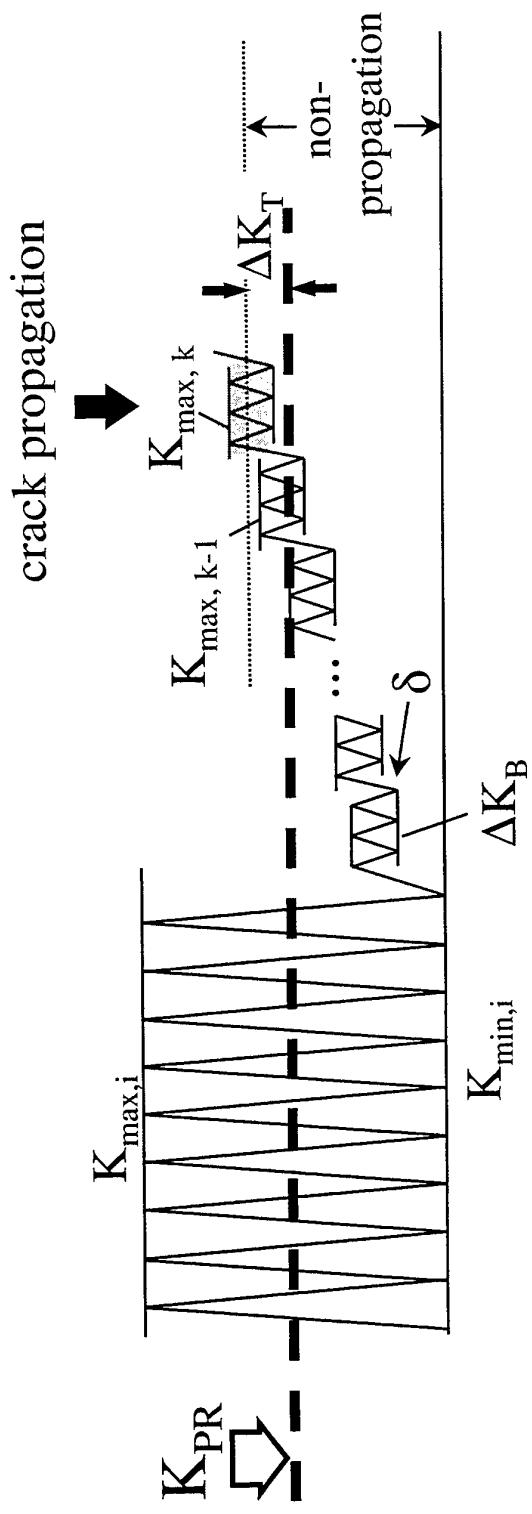


**FELIX (represents first 90 sec. Training)**

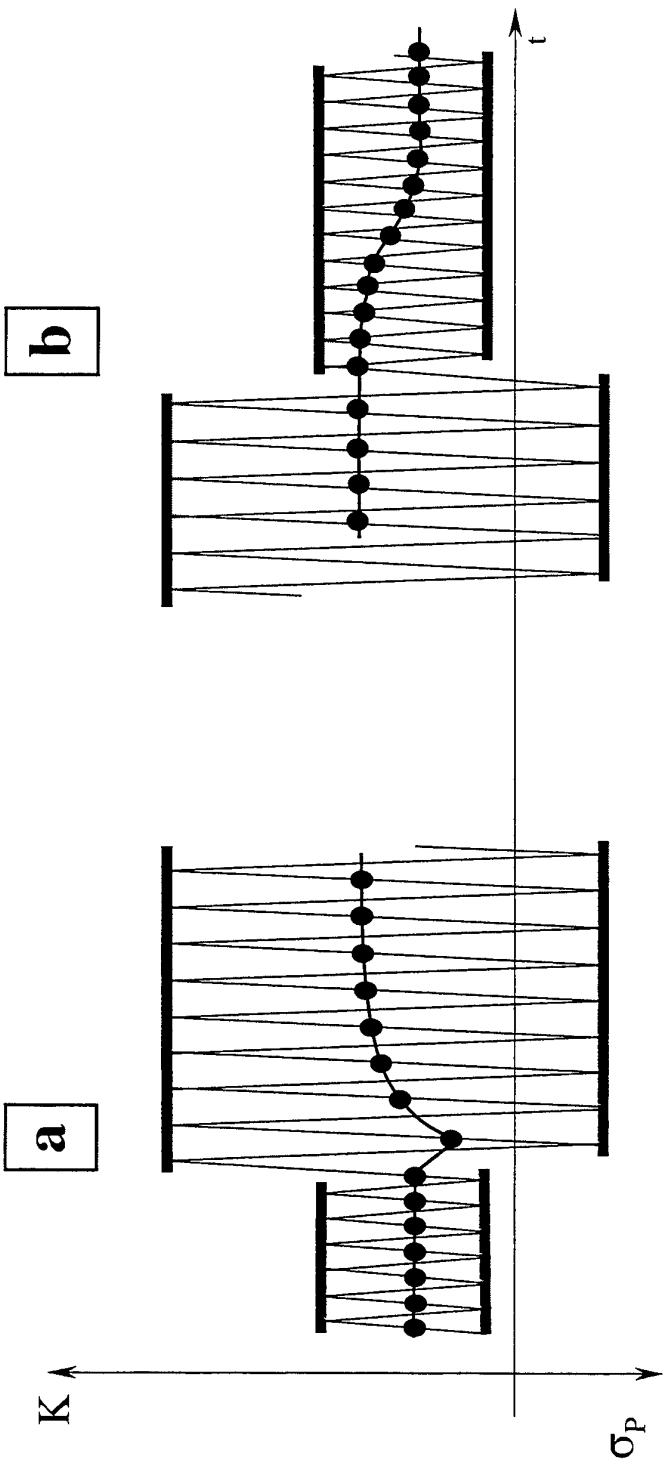


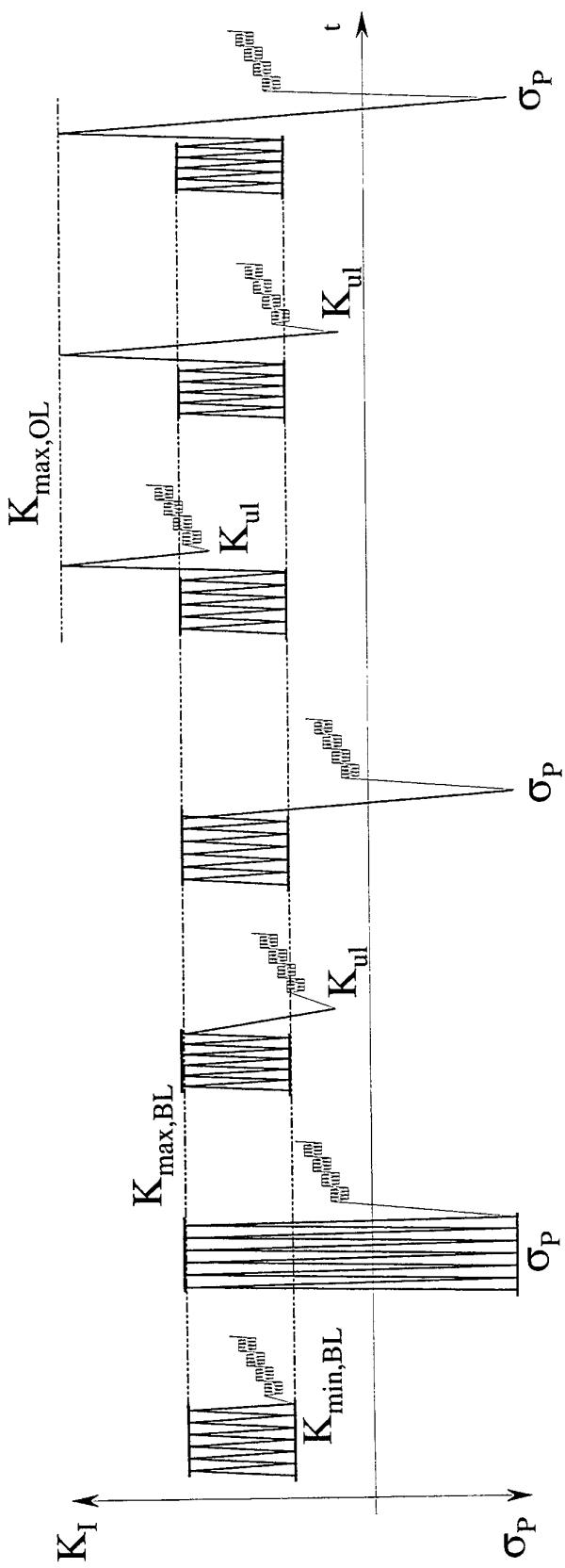


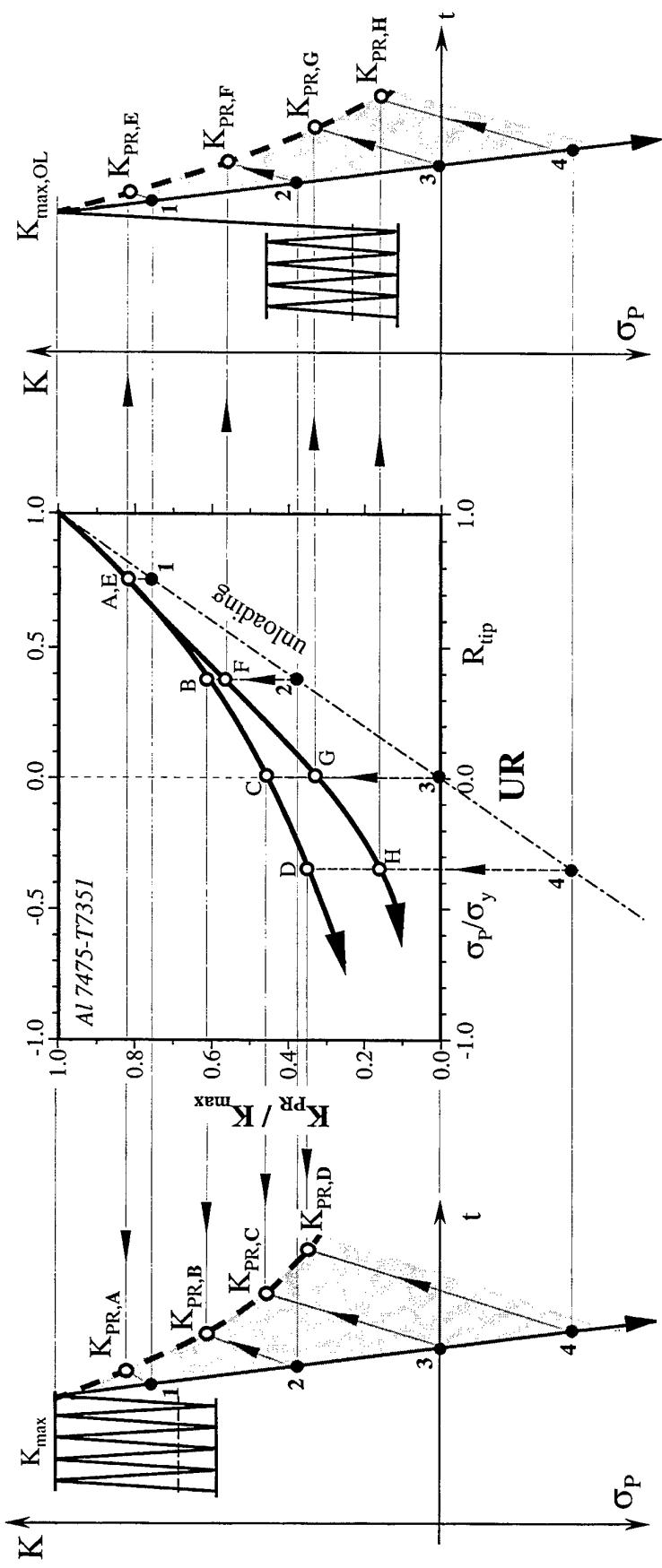
# Crack Propagation Load Measurement Method (CPLM)



$$K_{PR} = \frac{(K_{\text{max},k} + K_{\text{max},k-1})}{2} - \Delta K_T$$



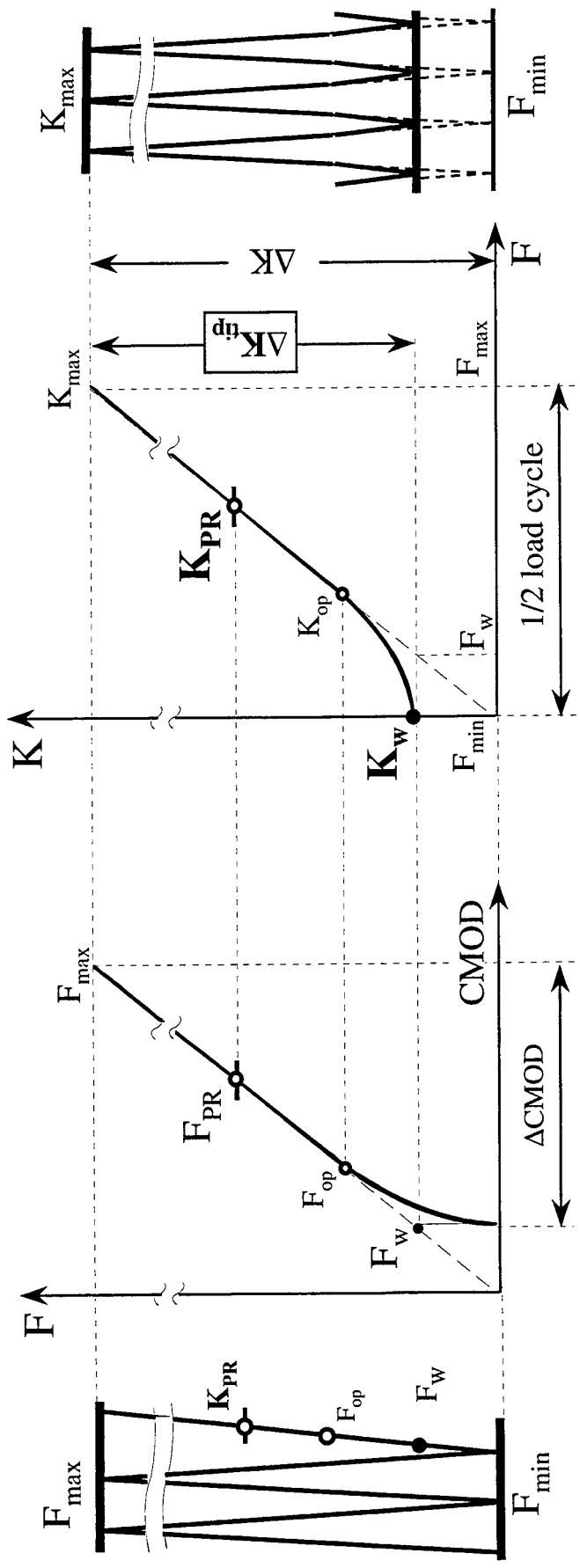




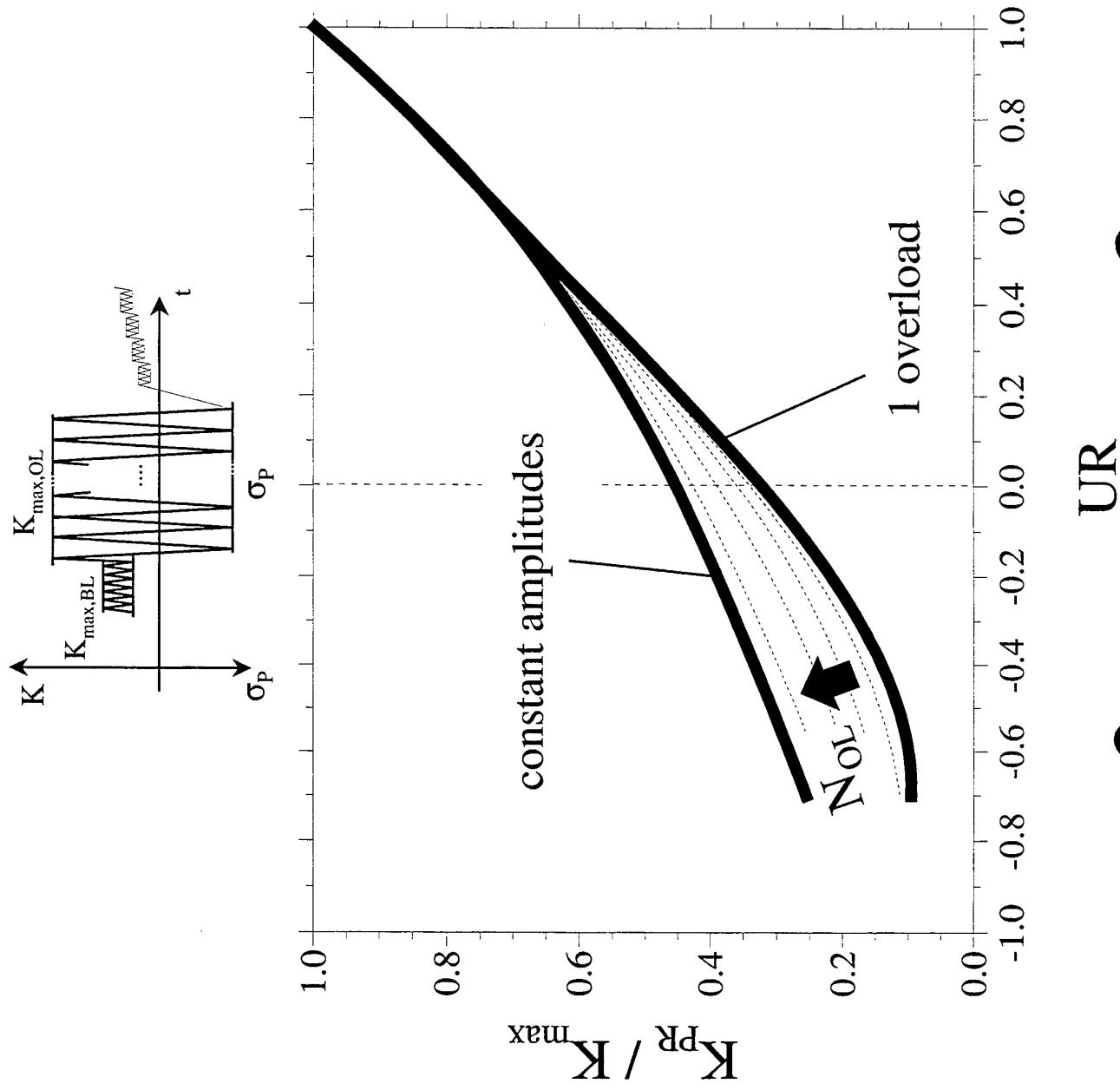
$$K_{PR,C} = \left( 0.453 + 0.34 \cdot UR + 0.134 \cdot UR^2 + 0.07 \cdot UR^3 \right) \cdot K_{max}$$

$$K_{PR,OL} = \left( 0.322 + 0.57 \cdot UR + 0.23 \cdot UR^2 - 0.145 \cdot UR^3 \right) \cdot K_{max,OL}$$

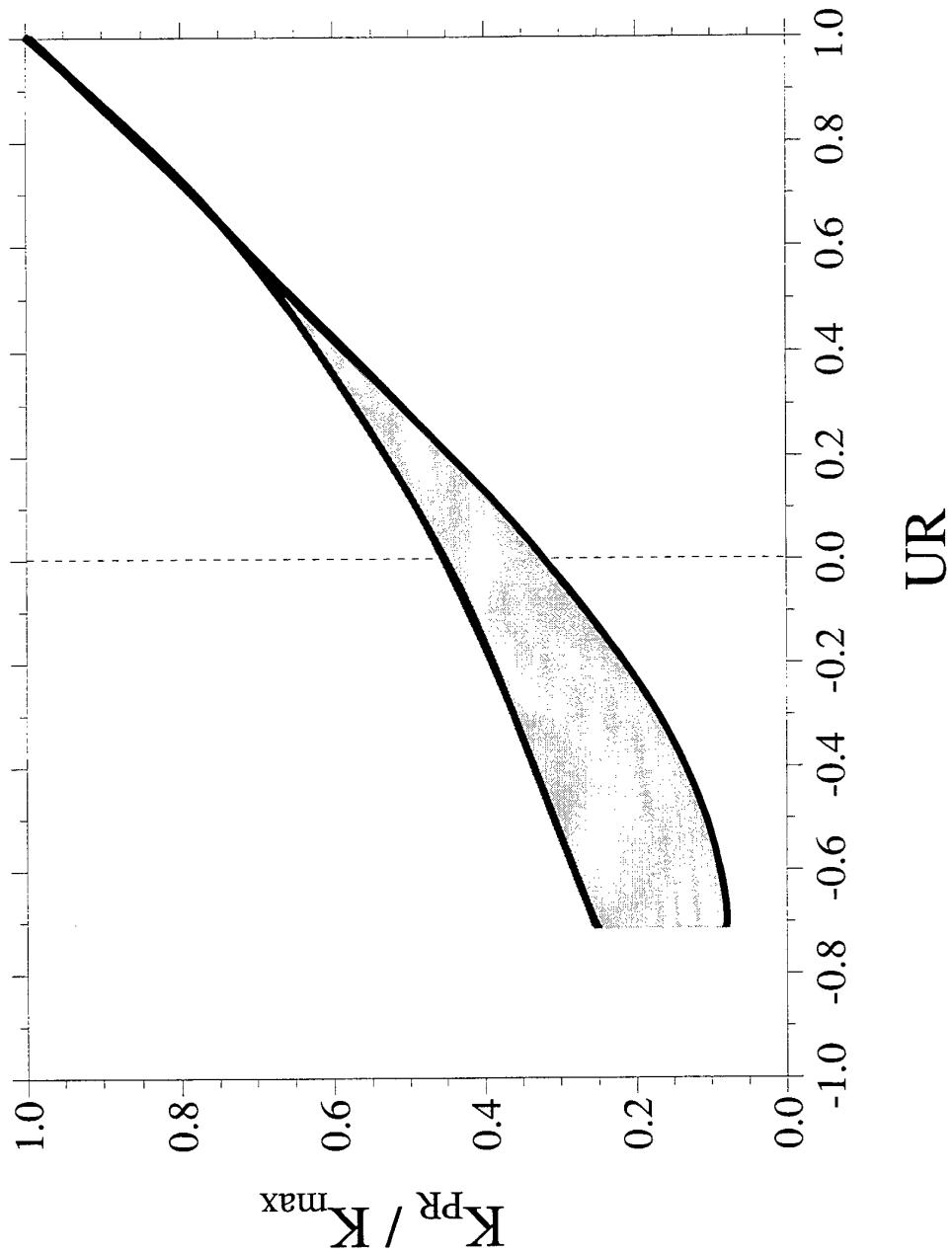
$$UR = R_{tip} = \frac{K_w}{K_{max}} \quad UR = \frac{\sigma_p}{\sigma_y}$$



$$\Delta K_{tip} = K_{max} - K_W$$

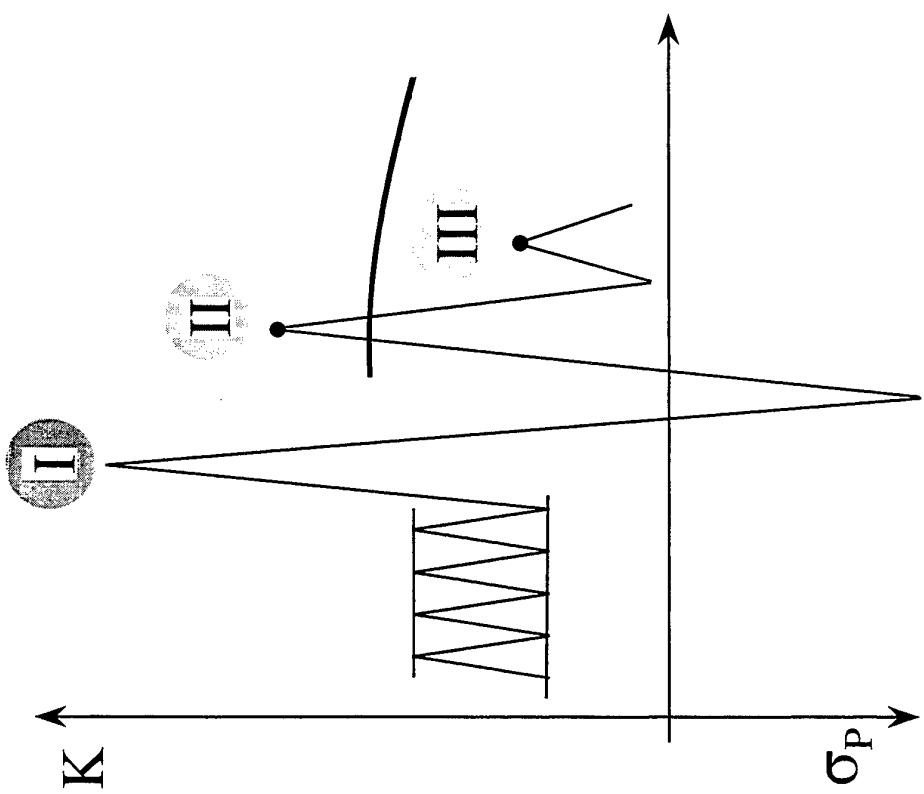


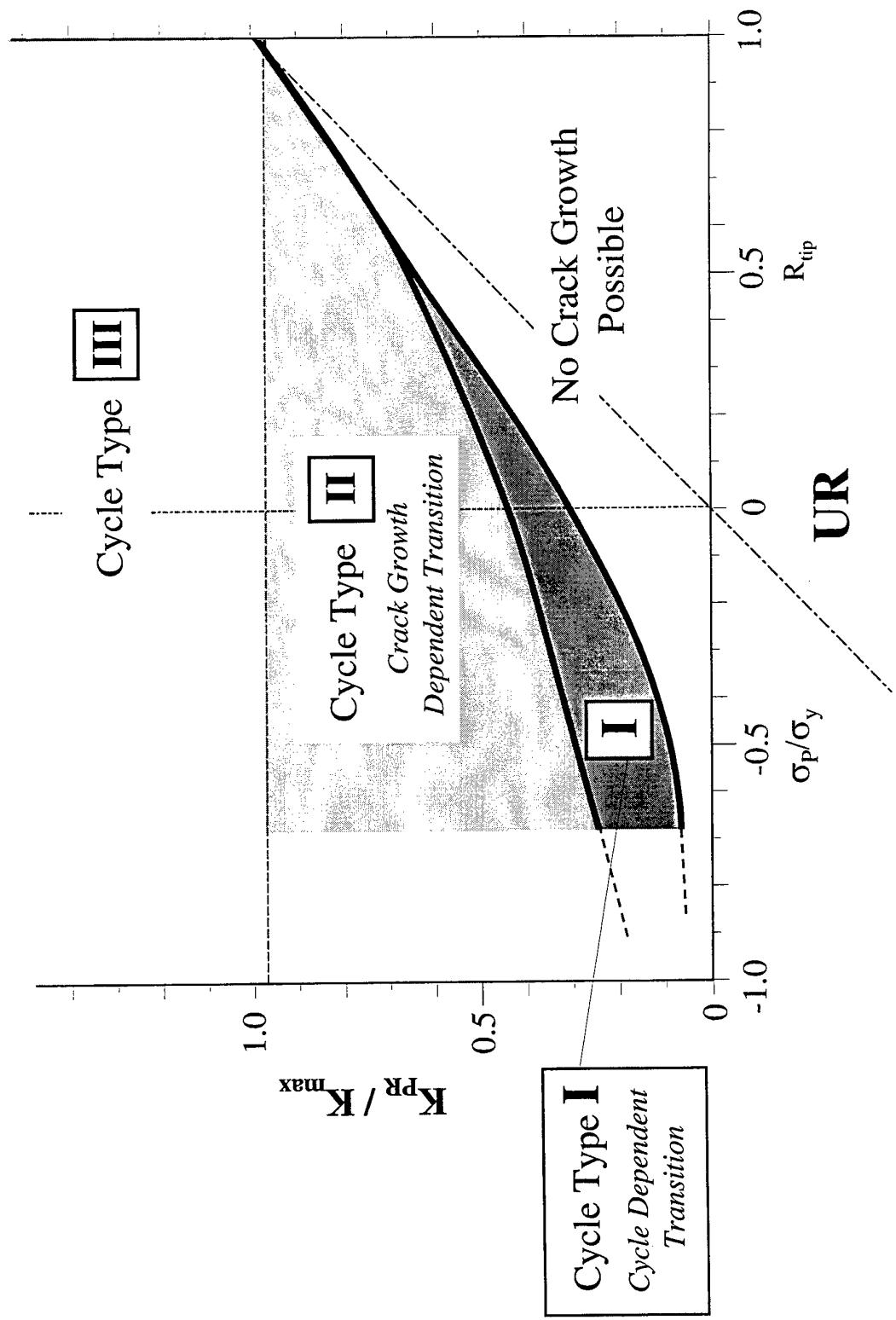
## Decisive Crack Propagation Conditions



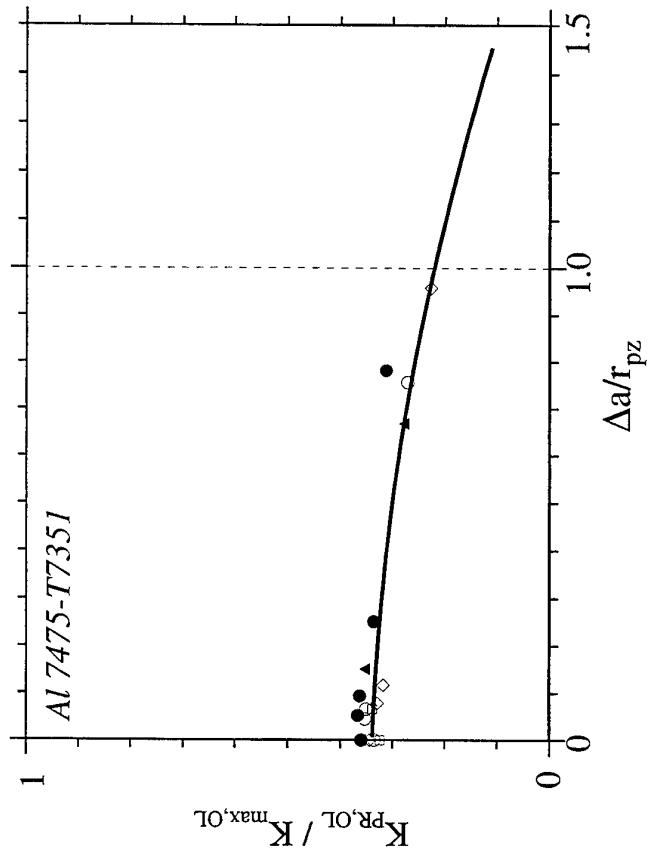
# The Three Possible Types of Cycles

- I  $K_{PR,OL} \leq (K_{PR} / K_{\max})_I \leq K_{PR,C}$
- II  $K_{PR,C} \leq (K_{PR} / K_{\max})_{II} \leq 1$
- III  $(K_{PR} / K_{\max})_{III} > 1$



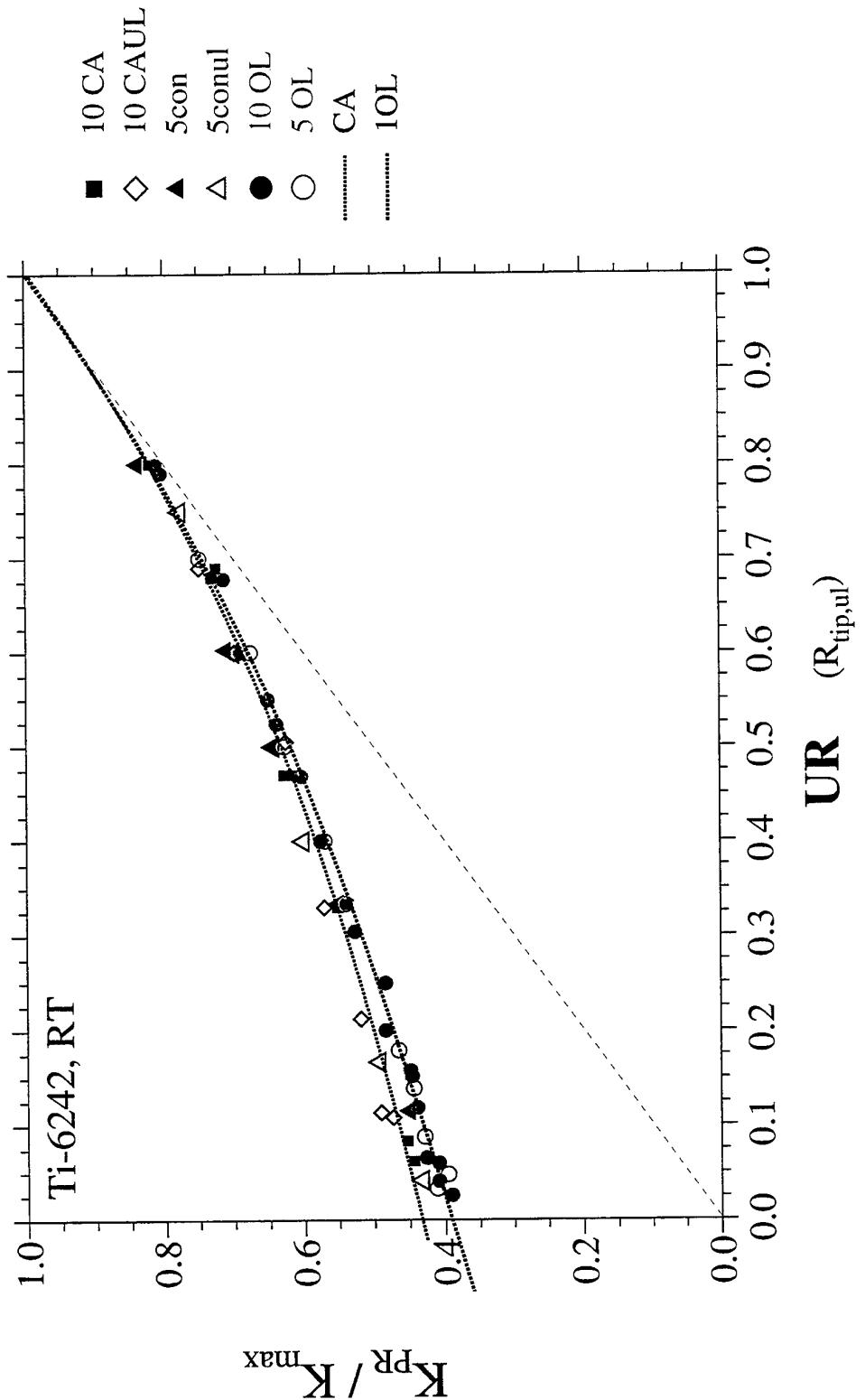


## Decline Curve of $K_{PR}$



$$K_{PR,decl.} = \left[ \frac{K_{PR,I}}{K_{max,I}} - \left( l \cdot \left( \frac{\Delta a}{r_{pz}} \right) + m \cdot \left( \frac{\Delta a}{r_{pz}} \right)^2 \right) \right] \cdot K_{max,I}$$

$$r_{pz} = \frac{2}{3\pi} \cdot \left( \frac{K_{max} - K_w}{\sigma_y} \right)^2$$

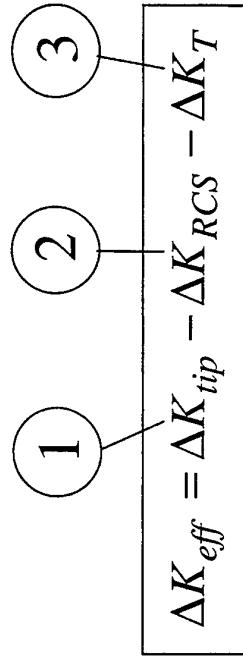


$$K_{PR,C} = \left( 0.433 + 0.311 \cdot UR + 0.09 \cdot UR^2 + 0.16 \cdot UR^3 \right) \cdot K_{max}$$

[0 ≤ UR ≤ 1]

$$K_{PR,1OL} = \left( 0.391 + 0.393 \cdot UR + 0.03 \cdot UR^2 + 0.184 \cdot UR^3 \right) \cdot K_{max,OL}$$

$$\Delta K_{eff} = K_{max} - K_{PR} - \Delta K_T$$



1.  $\Delta K_{tip}$  - Crack closure (crack face contact behind the crack tip) modifies the loading conditions by reducing  $\Delta K$  to  $\Delta K_{tip}$

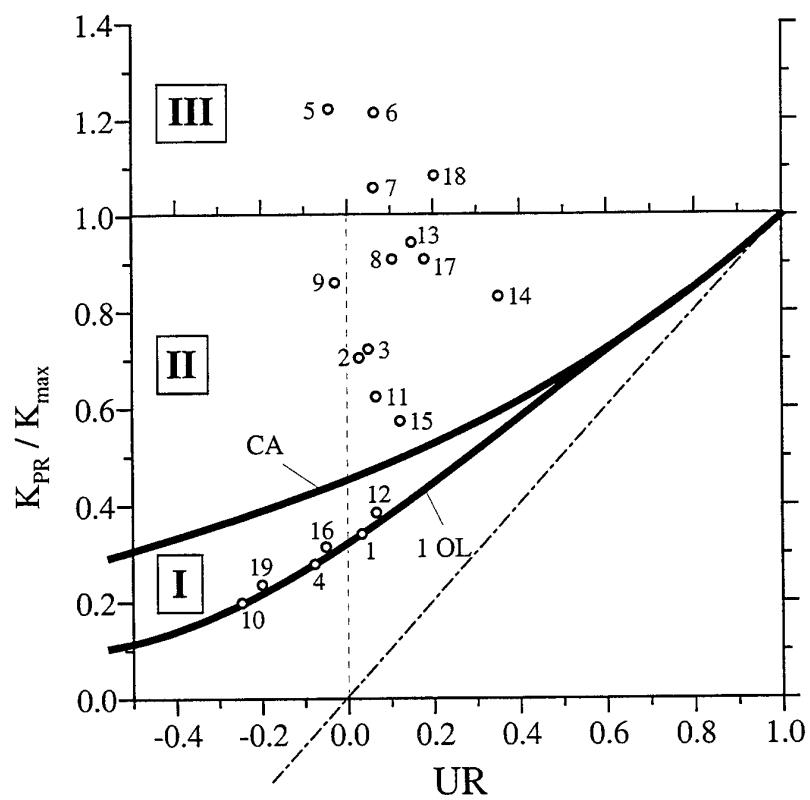
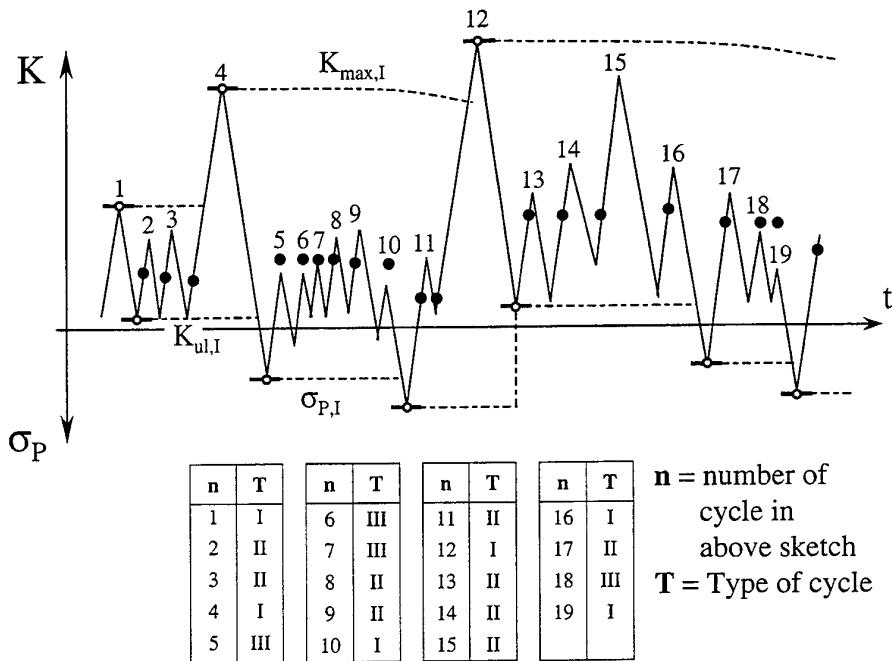
$$\Delta K_{tip} = K_{max} - K_w$$

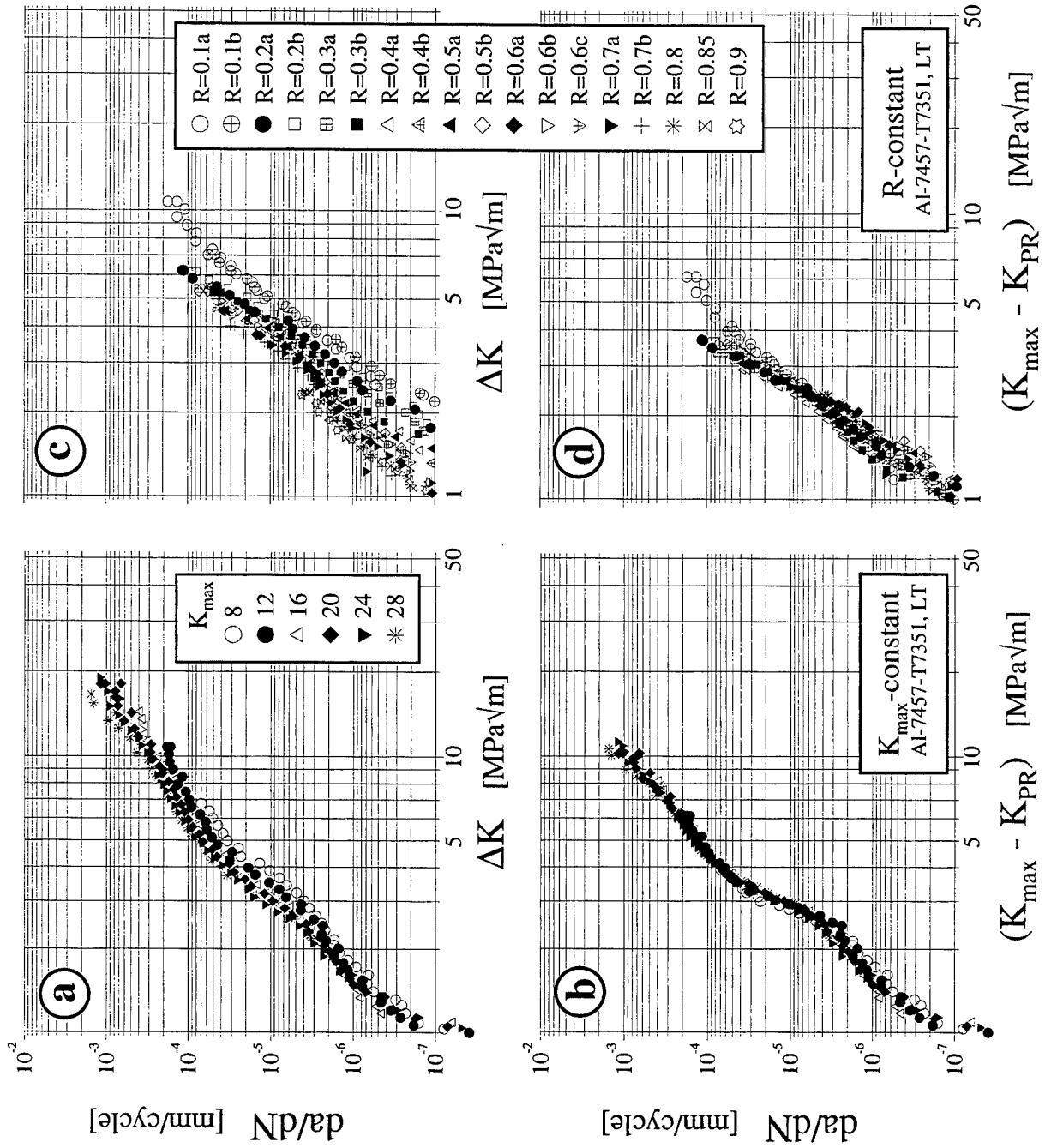
2.  $\Delta K_{RCS}$  - Local crack tip residual compressive stresses, accounted for by  $\Delta K_{RCS}$  contribute to the reduction from  $\Delta K$  to  $\Delta K_{eff}$ .

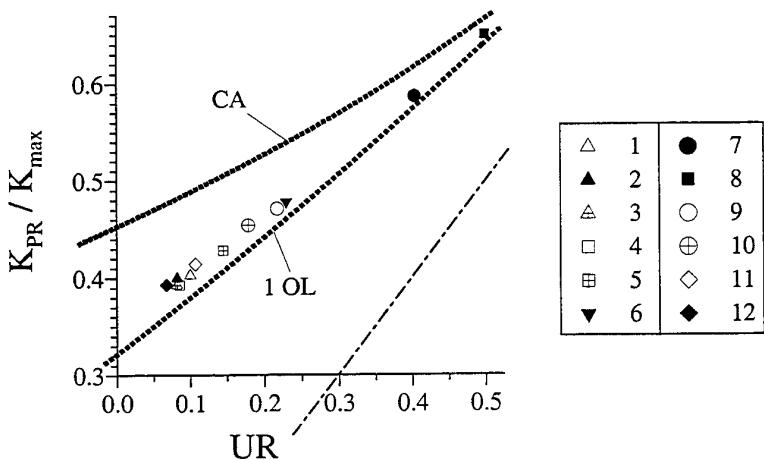
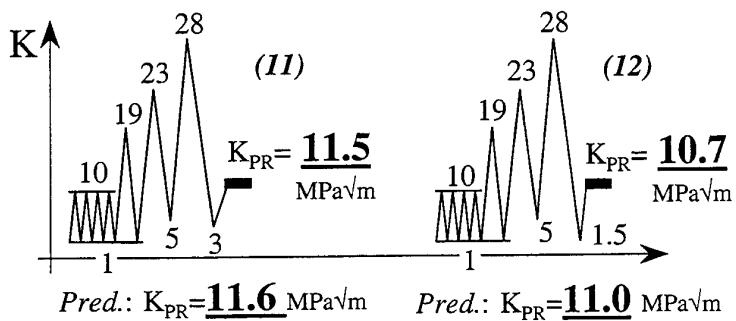
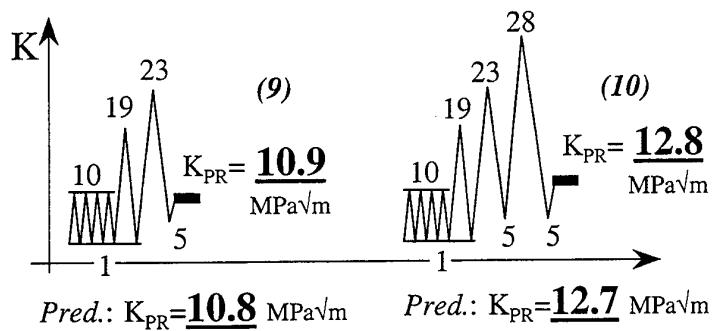
$$\begin{aligned} \Delta K_{RCS} &= K_{PR} - K_{min} && \text{(tension)} \\ &= K_{PR} - K_w && \text{(compression)} \\ \Delta K_{RCS} &= K_{PR} - K_w \end{aligned}$$

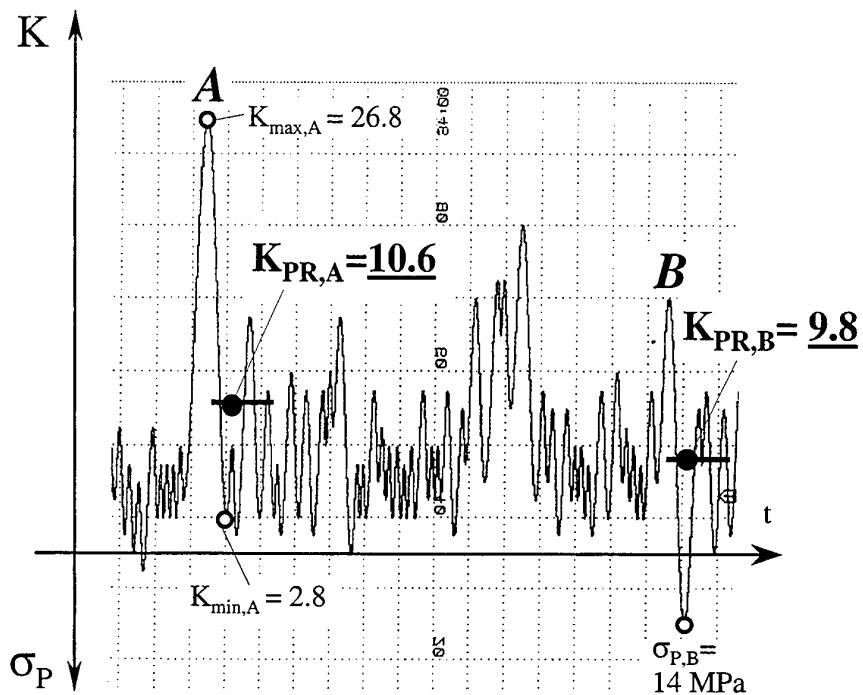
3.  $\Delta K_T$  - The material threshold value,  $\Delta K_T$ .

$$\Delta K_T$$









Calculation of  $K_{PR,A}$

$$K_{max,OL} = 26.8 \text{ MPa}\sqrt{\text{m}},$$

$$K_w = 2.8 \text{ MPa}\sqrt{\text{m}},$$

$$UR = R_{tip,ul} = 0.1045,$$

$$N_{OL} = 1.72,$$

$$K_{PR}/K_{max,OL} = 0.403,$$

$$K_{PR,A} = \underline{\underline{10.8}} \text{ MPa}\sqrt{\text{m}}$$

Calculation of  $K_{PR,B}$

$$UR = -0.03,$$

$$K_{max,OL} = 26.8 \text{ MPa}\sqrt{\text{m}},$$

$$N_{OL} = 1.72 \text{ (from cycle A)}$$

$$K_{PR}/K_{max,OL} = 0.331$$

$$K_{PR,B} = \underline{\underline{8.9}} \text{ MPa}\sqrt{\text{m}}$$

## Summary

- A new crack growth prediction model was presented
- Criterion is crack propagation ,  $K_{PR}$
- The model accounts for **crack closure, residual compressive stresses, intrinsic threshold value  $\Delta K_T$ .**
- Two type of transitions: **Cycle dependent and crack growth dependent transition**
- Only **three types of cycles** exist
- Necessary Input/Knowledge:
  1. Loading spectrum;  $K$  and  $\sigma_P$ .
  2. The correction of the loading conditions due to crack closure.  
(In many cases negligible).
  3. The knowledge of the **Fatigue Crack Growth Map** and the methodology.
  4. The definition of the driving force for fatigue crack growth given by

$$\Delta K_{eff} = K_{max} - K_{PR} - \Delta K_T$$

- 5. Material dependent parameters and functions which are:
  - 5.1. yield strength,  $\sigma_y$ ,
  - 5.2. the intrinsic threshold value,  $\Delta K_T$ ,
  - 5.3. the two “master curves” for CA and 1OL

$$K_{PR} = f(UR) \cdot K_{max}$$

- 5.4.the transitional function of  $K_{PR}$  between the two “master curves,”(**crack growth dependent transition**)
- 5.5 the decline function of  $K_{PR}$  (**cycle dependent transition**)
- 5.6 the function

$$da/dN = f(\Delta K_{eff})$$

to calculate the crack growth increment  $da$  per cycle.

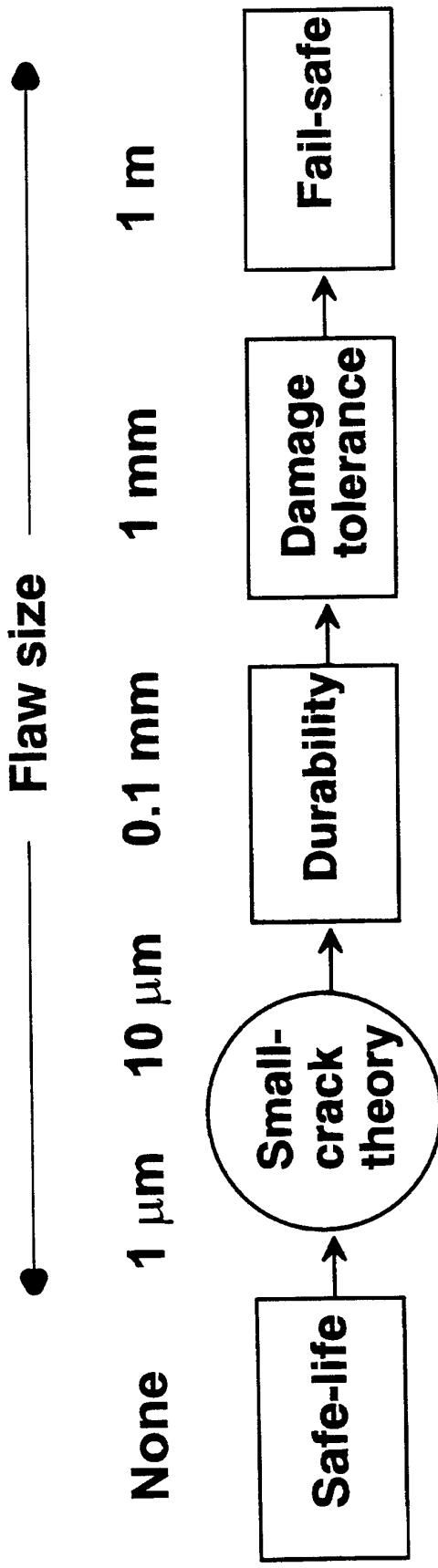
# FATIGUE LIFE ANALYSIS OF 4340 STEEL WITH A MACHINE-LIKE SCRATCH (Using Small-Crack Theory)

R.A. Everett, Jr.

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NASA Langley Research Center  
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Workshop on Damage Tolerance in Helicopters  
Cranfield University  
Cranfield, United Kingdom  
4-5, April 2000

# DESIGN CONCEPT USING SMALL-CRACK THEORY



- Hidden or uninspectable structure
- Manufacturing- and service-induced damage
- EIFS to fit S-N or  $\varepsilon$  - N behavior
- Economic-life extension
- Inspectable flaw size
- Redundant structure

# FAR 29.571

## (b) Fatigue tolerance evaluation(*including tolerance to flaws*)

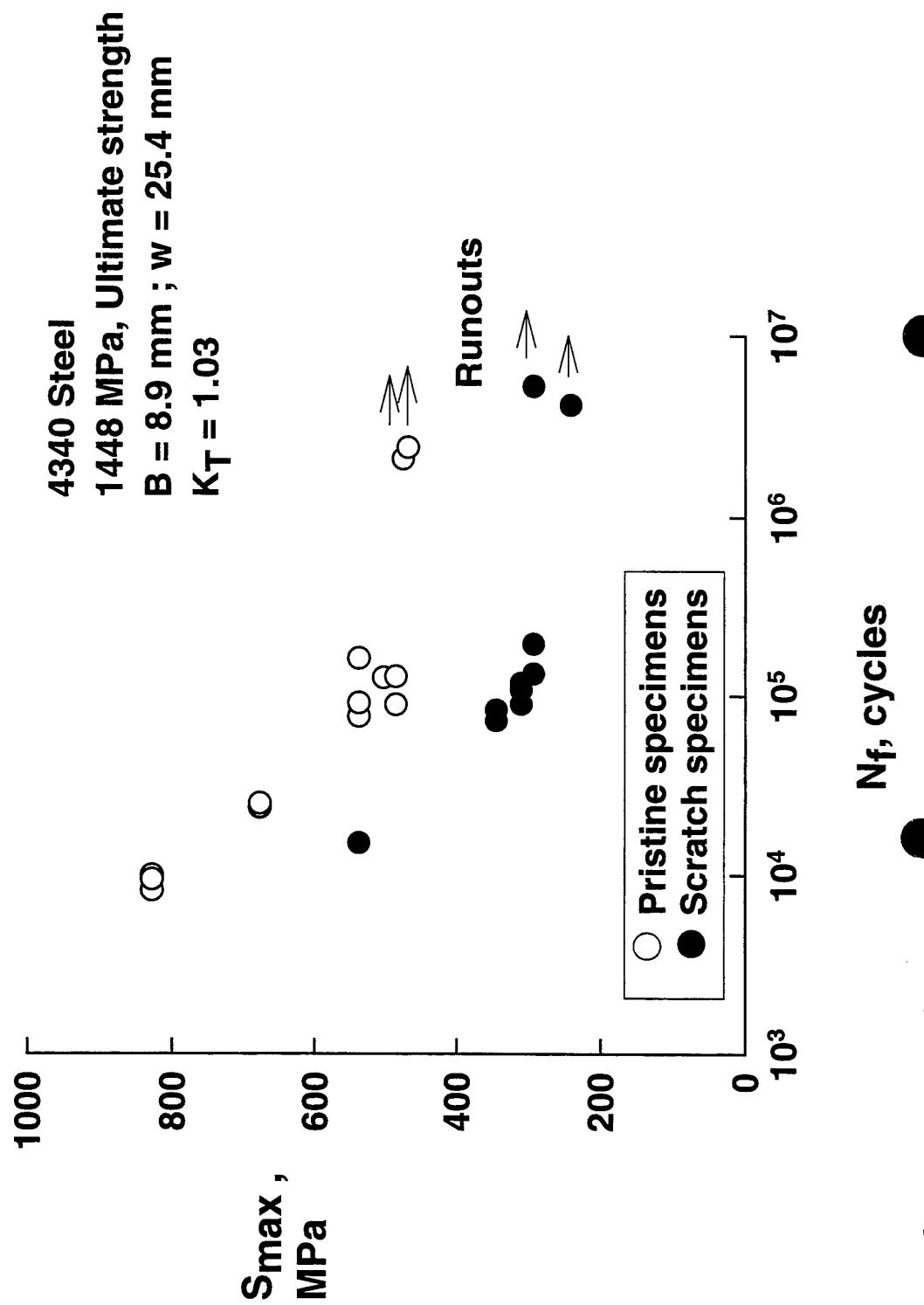
(1) *Flaw tolerant safe-life evaluation.* It must be shown that the structure, with flaws present, is able to withstand repeated loads of variable magnitude without detectable flaw growth for the following time intervals ---



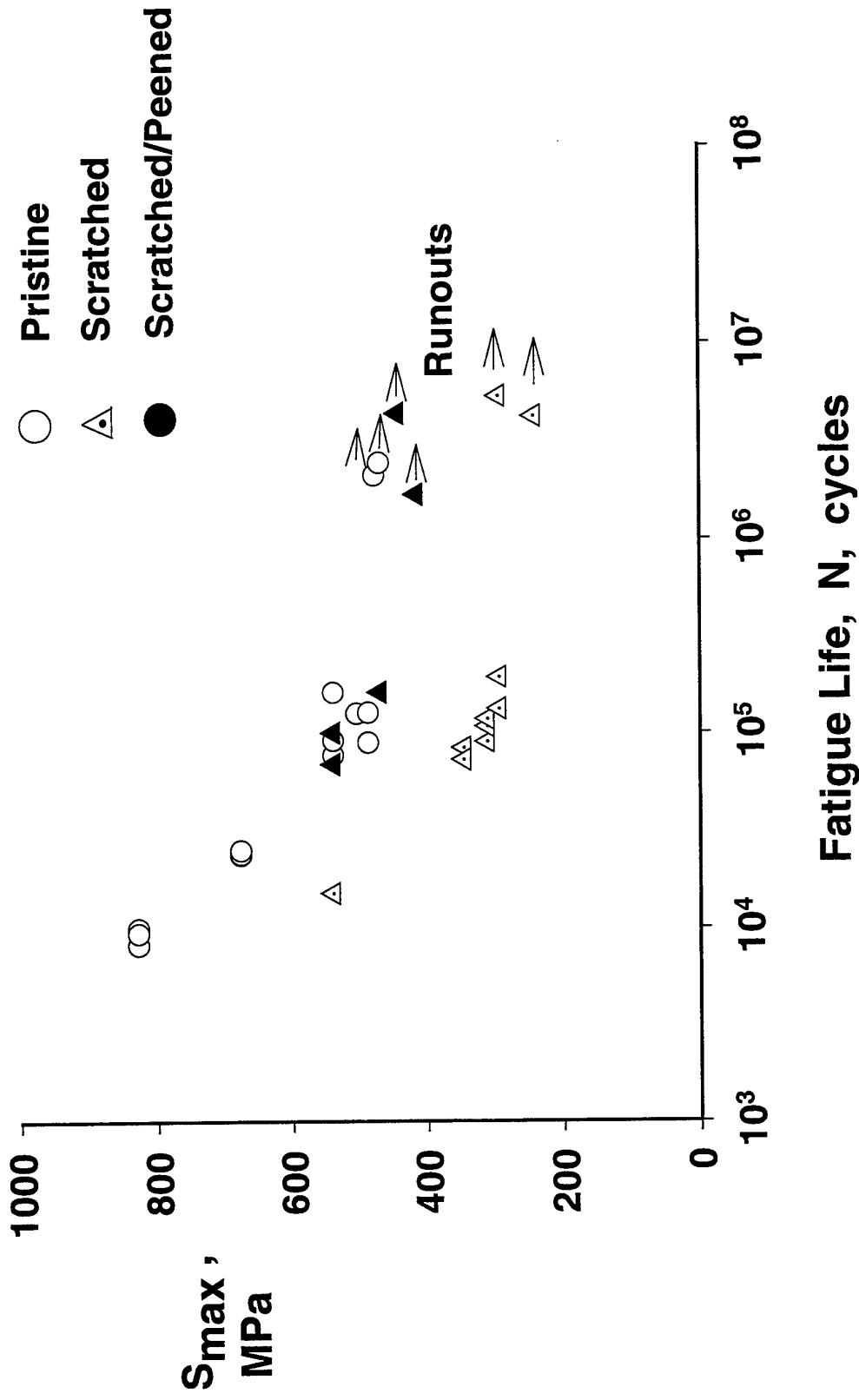
Fig. 6. Cross section of scratch with  
two small cracks (400X)

# FATIGUE LIFE OF PRISTINE AND SCRATCH SPECIMENS (SCRATCH DEPTH = 0.05 mm)

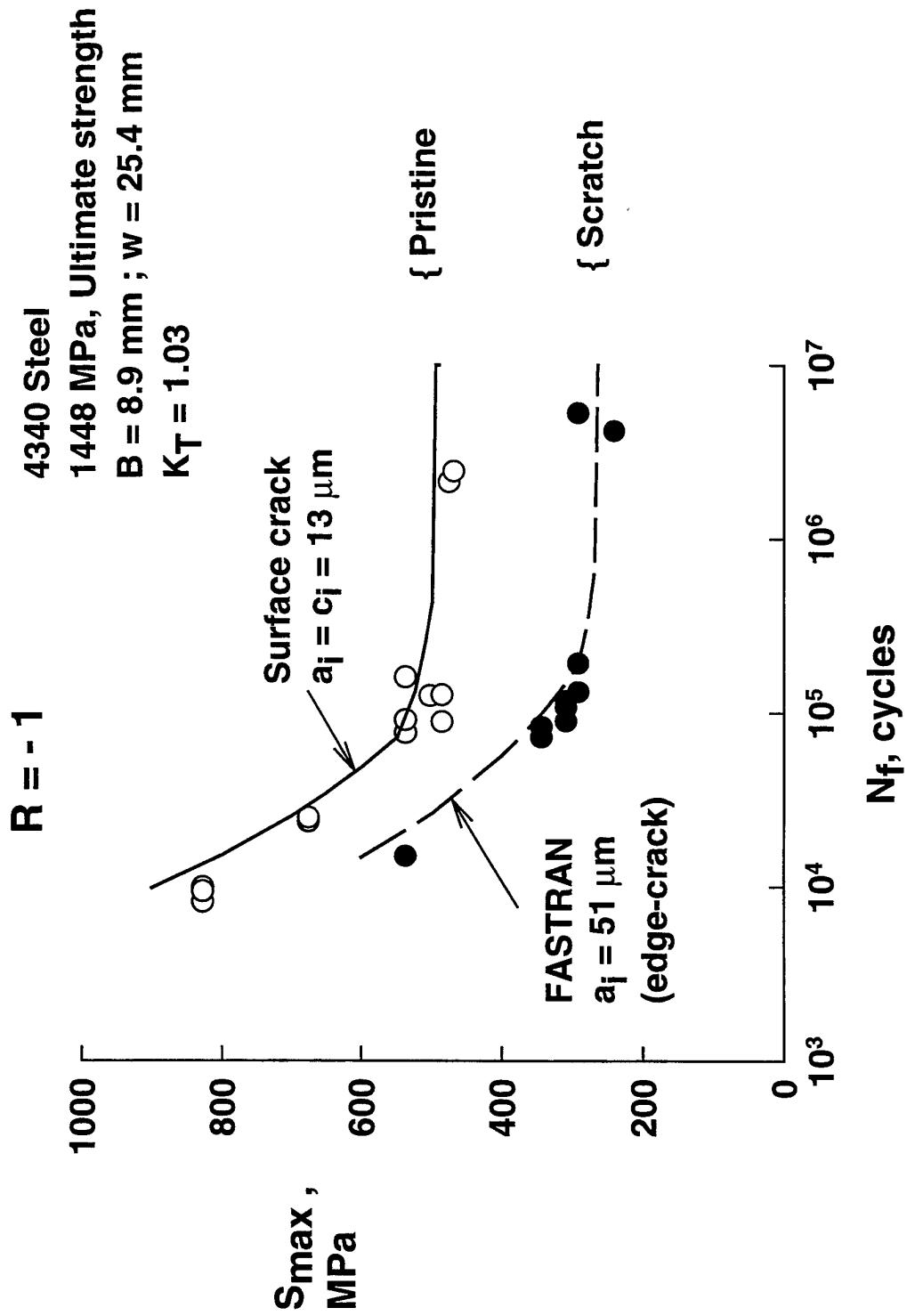
R = -1



# FATIGUE LIFE OF PRISTINE AND SCRATCH/PEELED SPECIMENS

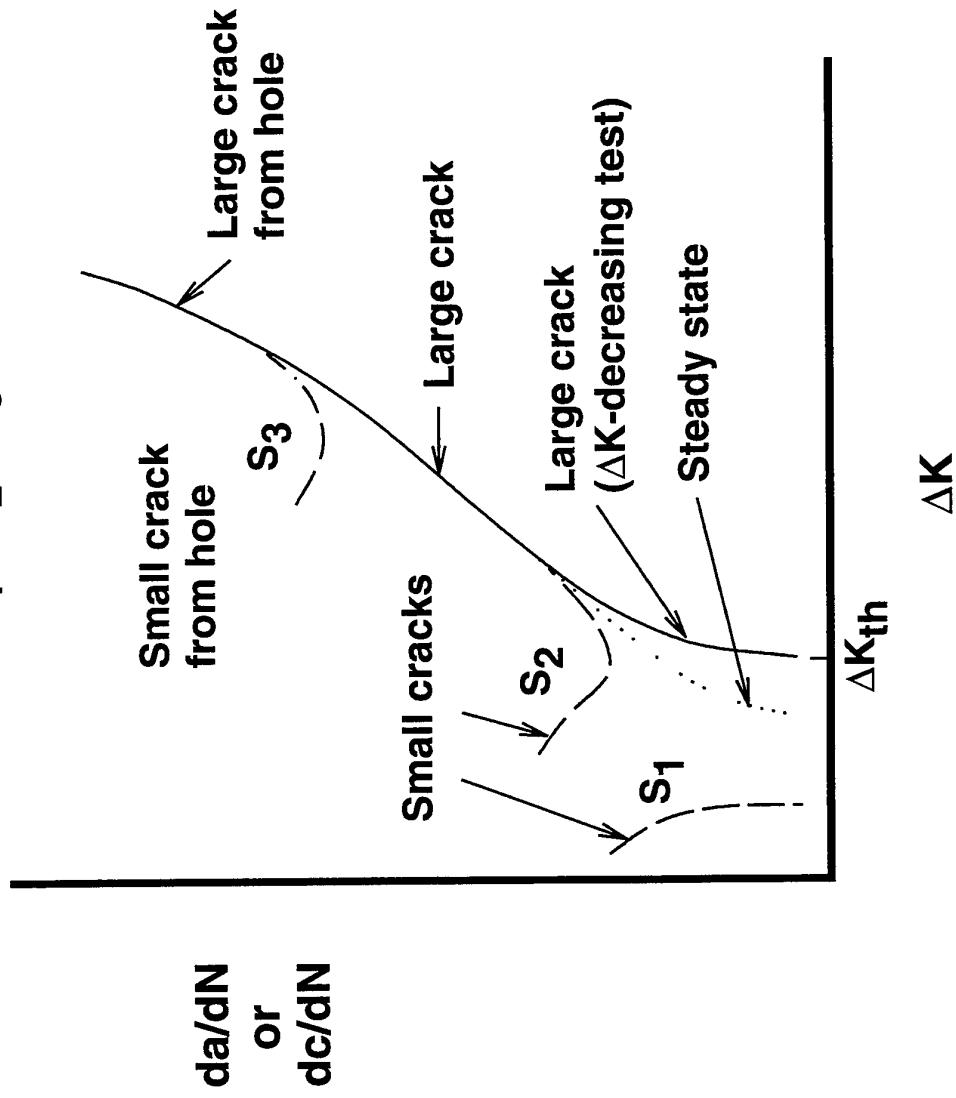


**MEASURED AND PREDICTED FATIGUE LIVES FOR  
THE PRISTINE AND SCRATCHED SPECIMENS**  
( Scratch Depth = 0.05 mm )

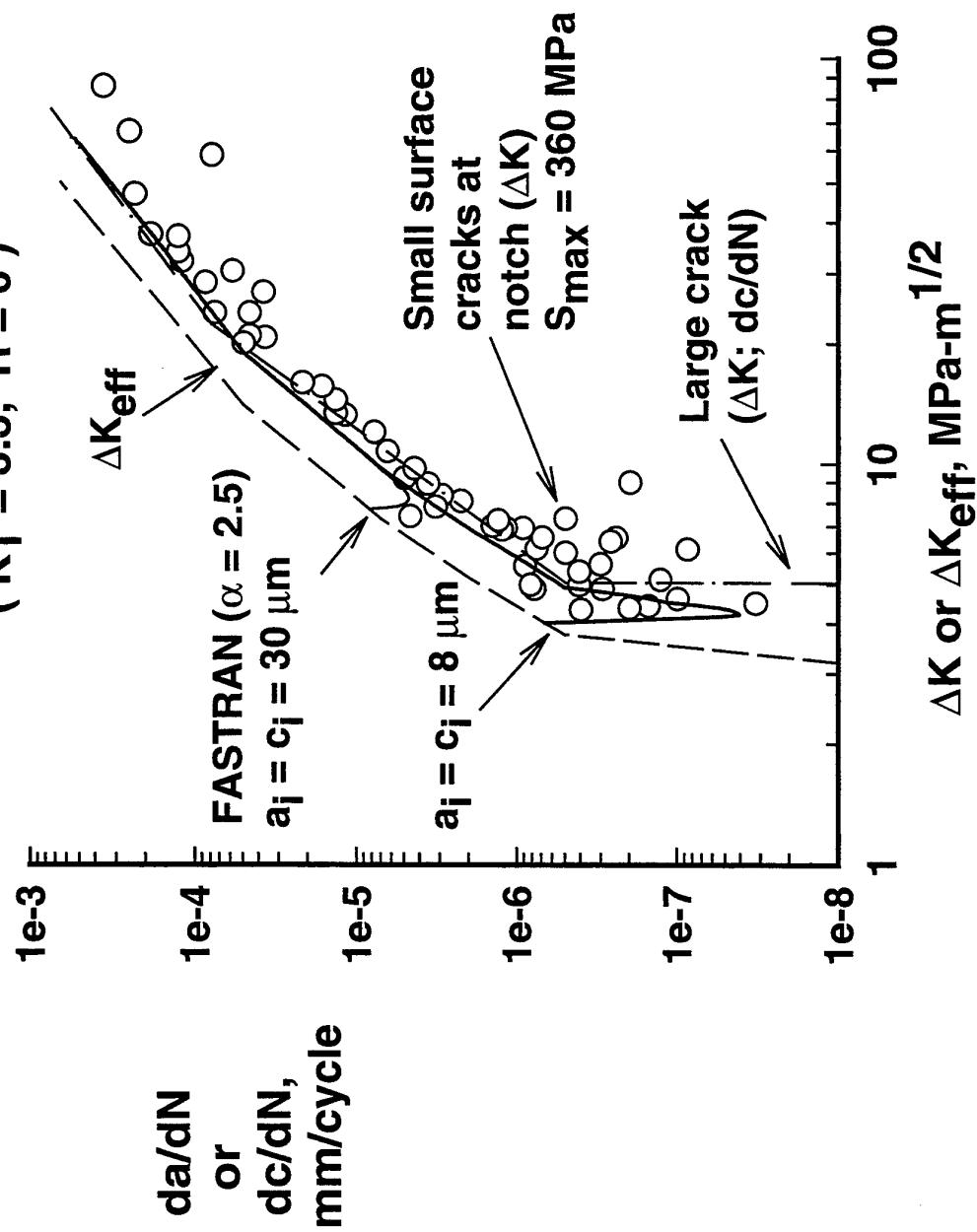


# THE "SMALL-CRACK" EFFECT

$$R = \text{constant}$$
$$S_1 < S_2 < S_3$$



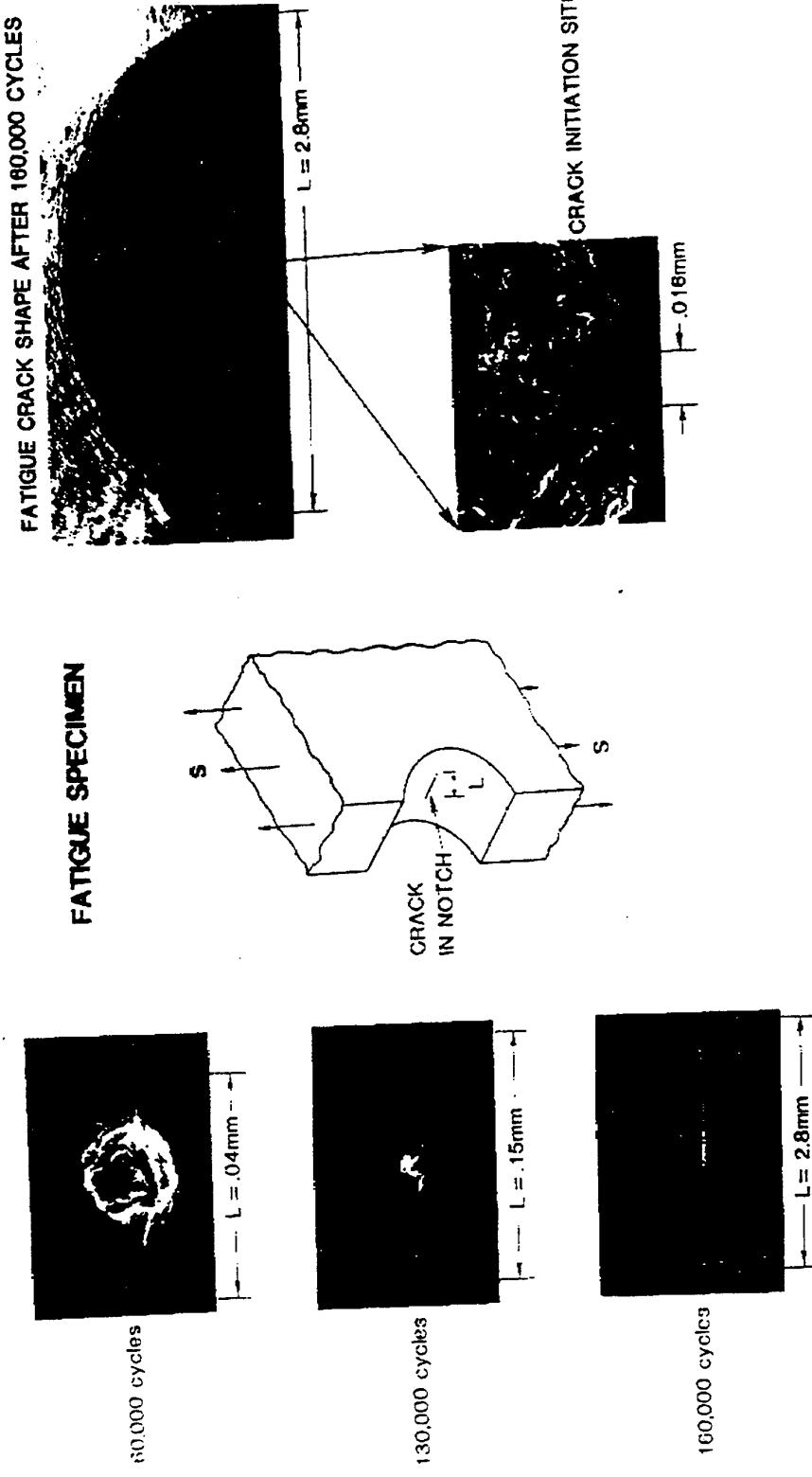
**MEASURED AND PREDICTED SMALL CORNER CRACK  
GROWTH AT A NOTCH IN 4340 STEEL**  
 $(K_T = 3.3, R = 0)$



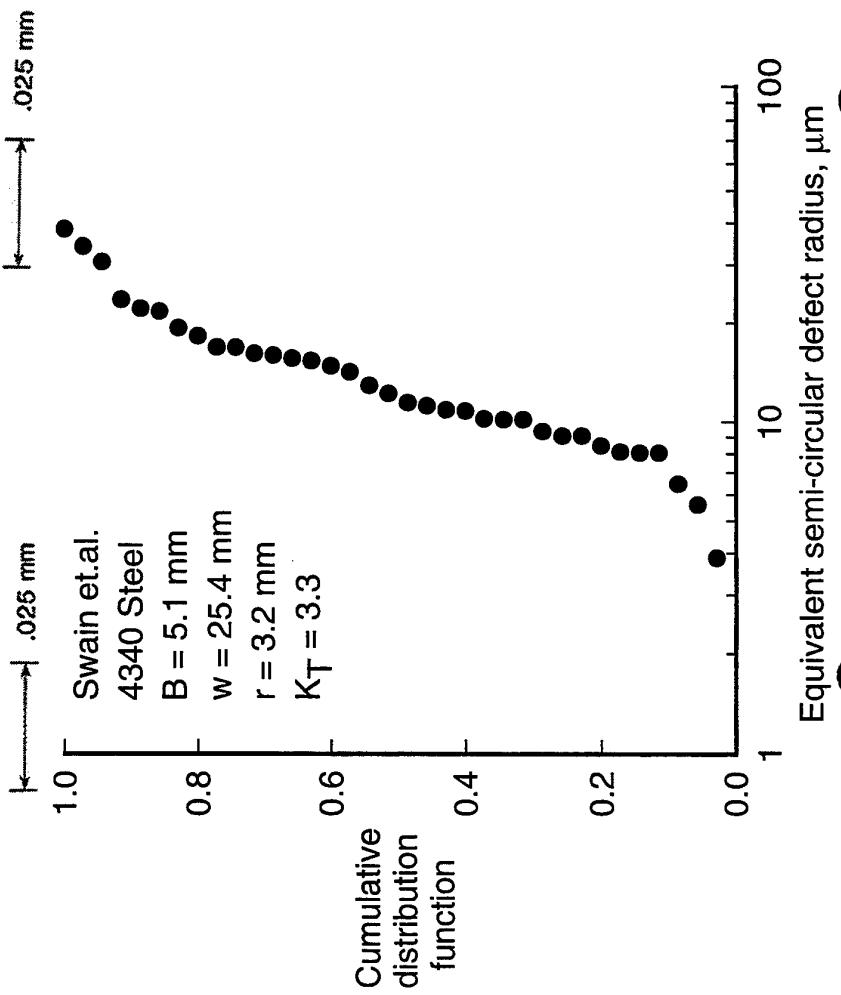
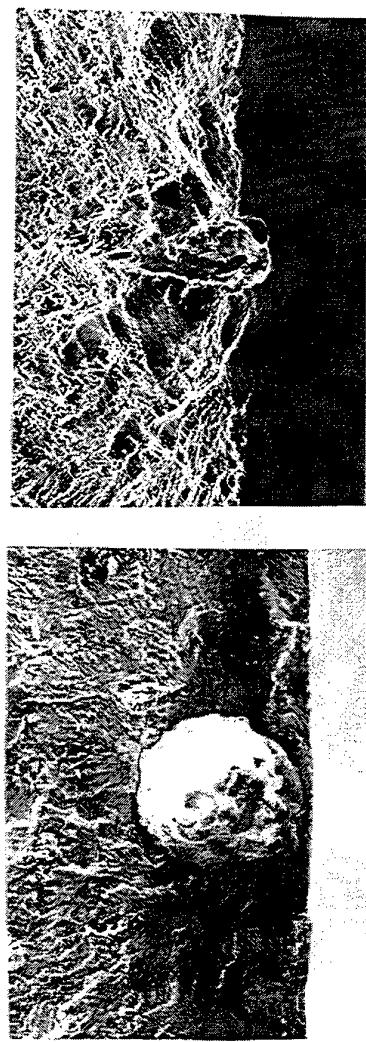
**AGARD-ARMY SUPPLEMENTAL SHORT CRACK PROGRAM  
ON EDGE-NOTCHED 4340 ALLOY STEEL**

**CRACK GROWTH MEASUREMENT  
ON NOTCH SURFACE FROM REPLICAS**

**FRACTURE SURFACE OF SPECIMEN**

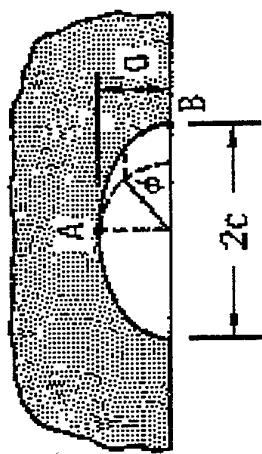


# Cumulative Distribution Function for Initiation Sites in 4340 Steel

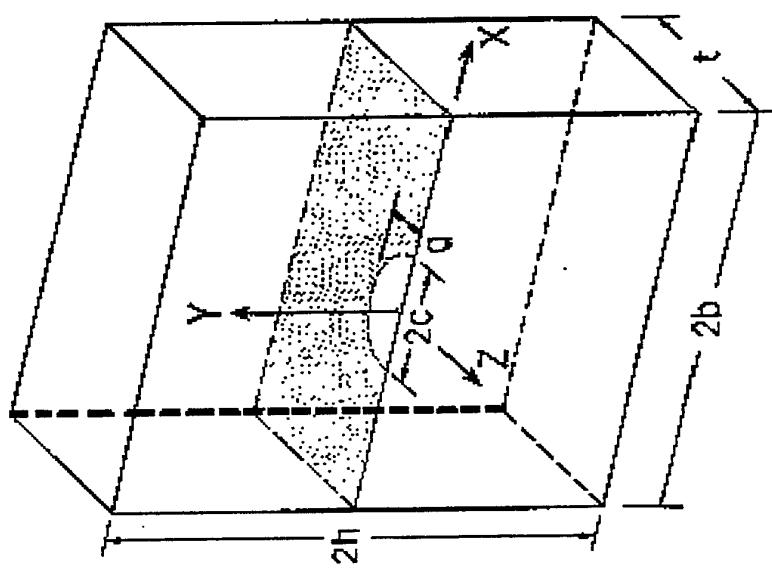


# Analytical & Weight Function Methods

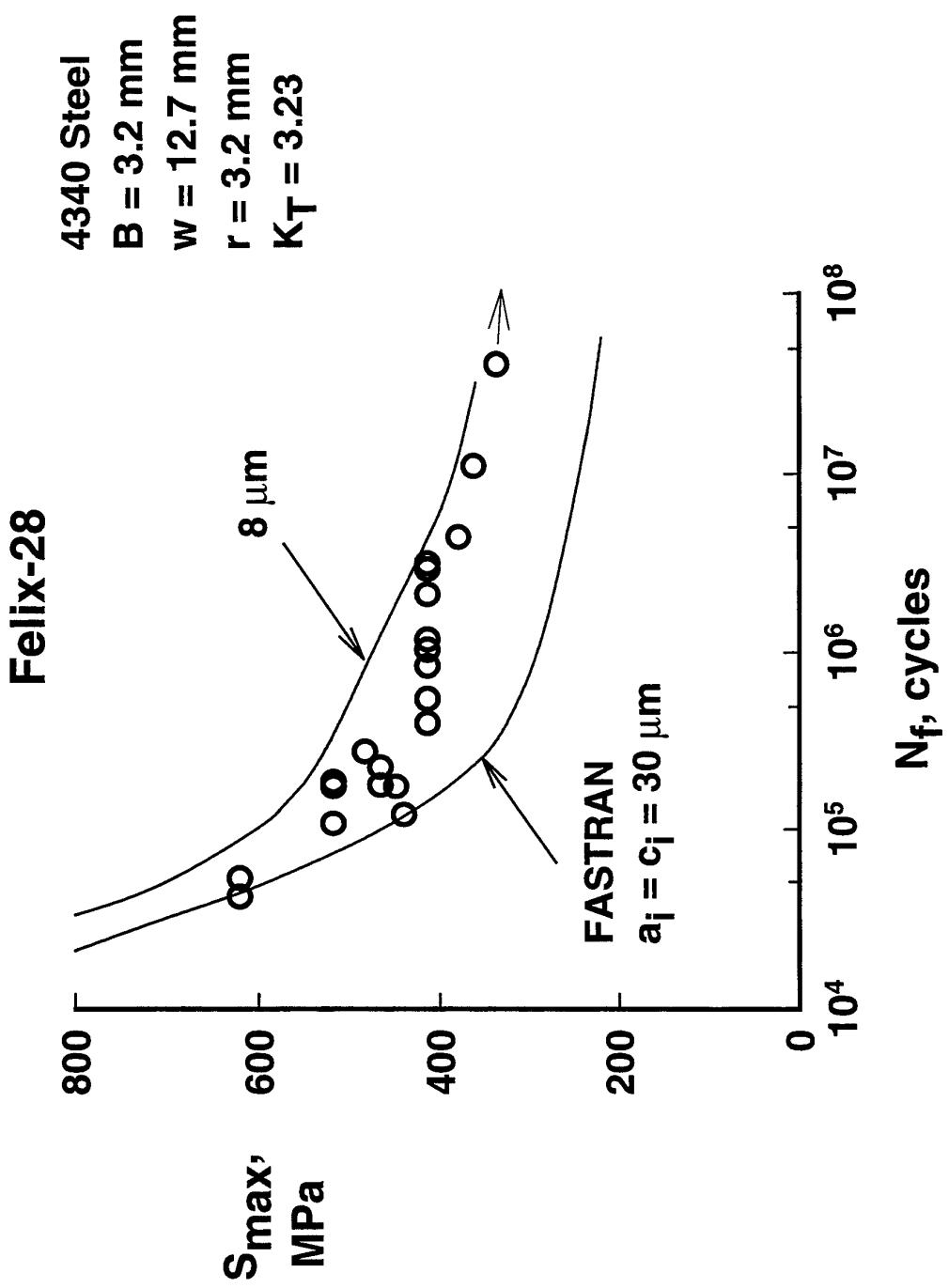
$$K = 1.12\sigma \sqrt{\frac{\pi a}{Q}}$$



$$\frac{da}{dN} = C (\Delta K)^n$$



# MEASURED AND PREDICTED FATIGUE LIVES UNDER SPECTRUM LOADING



## **CONCLUSIONS**

- A Machining-Like Scratch Reduced the Fatigue Strength of a High Strength By 40 %.
- Shot Peening Restored the Fatigue Strength.
- A Crack Growth Total Fatigue Life Model Predicted the S / N Behavior (Pristine and Scratched).
- A New Emerging Crack Growth Model for Predicting S / N Lives.

## **SESSION F**

# **DISCUSSION WORKSHOP: REGULATION ASPECTS**

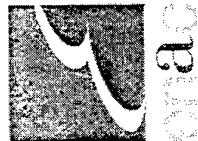


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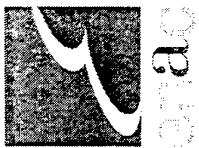


# DAMAGE TOLERANCE IN HELICOPTERS

## JAA Rulemaking and Advisory Material Development Initiatives, Harmonisation with FAA and their status

Workshop at Cranfield University - 4<sup>th</sup>, 5<sup>th</sup> April 2000

Bruno Moitre - ENAC, Aircraft Certification Division



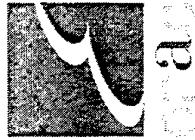
## JAA Rulemaking and Advisory Material Development Initiatives, Harmonisation with FAA and their status

- Introduction
- Regulatory initiatives in an FAA-JAA harmonised perspective
- The JAA Helicopter Airworthiness Study Group (HASG) and the Joint Harmonisation Working Group (JHWG)
- The Working Groups sponsored and their T.O.R.'s
- Concluding remarks



## Introduction

- Airworthiness design standards define minimum design requirements for ensuring a safety level congruent with the state of the art of the technology developments.
- Service experience, unairworthy conditions occurred should also be taken into account
- Emphasis put in the recent past on fatigue requirements and the associated need for continually review them by regulatory agencies are related to both events



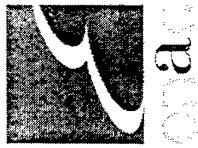
## Introduction ( cont . )

- The Technical Oversight Group for Ageing Aircraft ( TOGAA ) established in U.S.A. with their recommendations to the Rotorcraft Working Group and the Amendment 28 to FAR Part 29 are examples of how the said process develops under the said circumstances.



## Regulatory initiatives in an FAA-JAA harmonised perspective

- Nowaday a key word has entered into play within the regulatory civil aeronautical context :  
“Harmonisation”
- In such context Harmonisation has a meaning well beyond his literal one since the FAA and JAA committed in 1992 to develop a structure and formal procedure for their joint harmonisation program.
- The idea of harmonisation does not differ from the concept of defining a recognised and universally accepted standard however : .....



## Regulatory initiatives in an FAA-JAA harmonised perspective ( cont. )

### Why Harmonisation ?

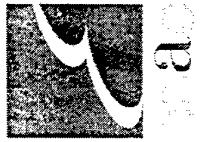
- The global market perspective
- The liberalisation of product's exchange
- The limit imposed to the barriers for air traffic growth
- Fair competition between U.S. and European Industry, Operators



## Regulatory initiatives in an FAA-JAA harmonised perspective ( cont. )

### What's Harmonisation?

- A fundamental process for the achievement of :
  - Common safety standards
  - Common interpretations of the requirements/means of compliance
- A challenge for the aviation community to achieve such goal while maintaining and possibly improving aviation safety.



## Regulatory initiatives in an FAA-JAA harmonised perspective ( cont. )

New regulations imply, in an *harmonised context* :

- A cost on the aviation community
- Costs have to be compensated on the ground of the mutual confidence and recognition of the Certificating Authorities
- Essential contributors for :
  - Certification cost reductions
  - Avoiding duplication of work
  - More efficient use of man power resources



The Helicopter Airworthiness Study Group  
and the Joint Harmonisation Working Group

The JAA Helicopter Airworthiness Study Group is  
the Authorities' group established by the JAA to

- develop harmonised proposals for JAR-27 and  
JAR-29 ( small and large rotorcraft respectively )
- support the maintenance of JAR-27, JAR-29

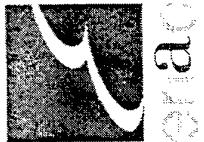


## The Helicopter Airworthiness Study Group and the Joint Harmonisation Working Group ( cont. )

The Joint Harmonisation Working Group ( JHWG ) is the joint Authorities' and Industries' group ( U.S. and European ) set up to cooperate for the development of harmonised airworthiness requirements for small and large rotorcraft .

The JHWG is composed of :

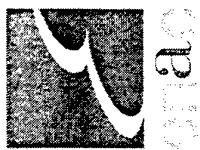
- the HASG members
- an FAA representative
- U.S. and European Industry representatives



## The Helicopter Airworthiness Study Group and the Joint Harmonisation Working Group ( cont. )

The JHWG is supported by ad hoc working groups composed of experts selected from Authorities and Industry when specific regulatory and interpretative issues need to be addressed in light of harmonisation.

Rulemaking initiatives agreed within the JHWG are approved by the Harmonisation Management Team to which proposals are addressed together with the T.O.R.'s envisaged.



The Working Groups sponsored by the JHWG  
and their T.O.R.'s

## Rulemaking

## Advisory Material

(existing req.'s)

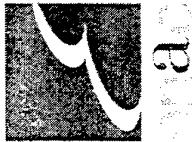




## The Working Groups sponsored by the JHWG and their T.O.R.'s ( cont. )

Four Working Groups for Fatigue of Rotorcraft Structures have been agreed to be formed

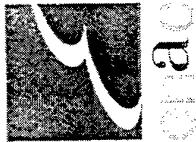
- W.G. C5 - D.T. and fatigue evaluation of metallic rotorcraft structure ( rulemaking )
- W.G. C8 - Fatigue evaluation of metallic rotorcraft structure ( A.C. updating )
- W.G. C6 - D.T. and fatigue evaluation of composite rotorcraft structure ( rulemaking )
- W.G. C7 - Fatigue evaluation of composite rotorcraft structure ( A.C. updating )



## T.O.R.'s for W. G. C5

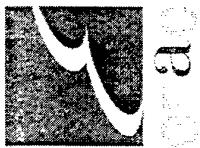
### BACKGROUND

- The Technical Oversight Group for Aging Aircraft (TOGAA) has recommended that current safe-life methods be complemented by D.T. assessment methods and that flaw tolerance safe life be removed from the regulations.
- The Industry (AECMA and AIA) in the White Paper, recommends retention of Flaw Tolerance as a Damage Tolerance option, and that the overall fatigue methodology be improved and updated.
- A Road Map is developed in the White Paper
- There is agreement that use of D.T. for part 27 will be optional



## SPECIFIC TASKS

- Evaluate the Industry White Paper, the recommendations contained in the TOGAA letters to the FAA, and the ongoing activities and results of rotorcraft D.T.R&D.
- Identify the information needed to conclude rulemaking and define acceptable M.O.C.
- Recommend appropriate changes to FAR/JAR 29 regarding D.T. and fatigue evaluation of metallic structure, and to FAR/JAR 27 ( D.T. as an option)



T.O.R.'s for W. G. C5 ( cont. )

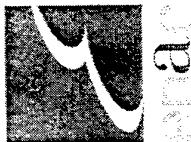
### SPECIFIC TASKS ( cont. )

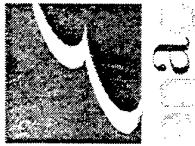
- Any recommended change should be practical/appropriate to the unique characteristics of rotorcraft.
- Where feasible and appropriate, provide consistency with FAR/JAR 23/25
- Prepare related AC material for both FAR/JAR 27 and 29.
- If requested, recommend disposition of comments received in response to the NPRM or NPA.

T.O.R.'s for W. G. C5 ( cont. )

## SCOPE/EXPECTATION

- The project is to be a harmonized JAR/FAR 27/29 ARAC program.
- The ARAC working group will present its final document to ARAC by April 2002
- Four meetings are anticipated, beginning April 2000
  - A progress report will be provided at each JHWG meeting
- The ARAC recommendation will be provided to the FAA and the JAA by September 2002.





## T.O.R.'s for W.G.C5 ( cont. )

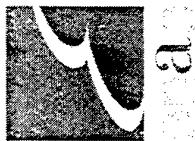
### CAVEAT

- A separate rotorcraft ARAC working group will be developing a rulemaking proposal for composite structure.
- Although this tasking for metallic structure does not depend on the completion of the composite structure project, the two working groups should communicate to avoid possibly conflicting recommendations to amend the same regulatory sections



## BACKGROUND

- Current FAR/JAR 27/29 regulations do not provide adequate certification standards for composite materials.
- Past certifications based on advisory material and a broad interpretation of the fatigue substantiation.
- Some AA's have issued Special Conditions because the advisory material is not supported by an adequate airworthiness standard.



## T.O.R.'s for W. G. C6 ( cont. )

### SPECIFIC TASKS

- Revise current FAR/JAR 27 and 29 to add regulations for composite structure
- Consider creating a new FAR/JAR 27/29.573.
- Evaluate and revise, as appropriate, the regulations-advisory material to achieve the goal of improved tolerance to flaws and defects in composite structure with methodology and procedures which are practical and appropriate to rotorcraft.
- Consistency with FAR/JAR 23/25 where feasible-appropriate

T.O.R.'s for W. G. C6 ( cont. )

## SCOPE/EXPECTATIONS

- The project is to be a harmonized JAR/FAR 27/29 ARAC program.
- The ARAC working group will present its final document to ARAC by June 2002.
- Four working group meetings are anticipated, beginning June 2000.
- A progress report will be provided at each JHWG meeting.



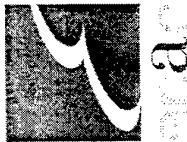


## T.O.R.'s for W.G. C6 ( cont. )

### CAVEAT

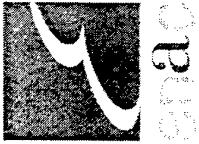
- A separate rotorcraft harmonisation W.G. will be developing a D.T. rulemaking proposal for metallic structure.
- Although this tasking for composite structure does not depend on the completion of the metallic structure project, the two W.G.'s should communicate to avoid possibly conflicting recommendations to amend the same regulatory sections.

## Concluding remarks ( cont. )



### On the reason for the 4 distinct W.G.'s sponsored

- Existing FAR/JAR 27,29 requirements for fatigue evaluation of rotorcraft structures are in the TC basis of a number of projects
- FAA Rotorcraft Directorate has mandate for revising and maintaining AC 27-1 and AC 29-2
- This task is promoted by the FAA Rot.Dir. through a two years revision cycle which forms also part of the JHWG routine work for the seeking of harmonisation
- WG's for AC revision are composed by the same people involved in the WG's for rulemaking thus allowing consistency, synergistic efforts, cost saving,



## Concluding remarks ( cont. )

### On the Road Map proposed by Industry

- R & D effort should parallel and support but not drive the rulemaking effort,to enhance the advisory material and expand the practical application of a very general proposed regulation.
- The AA's do not consider appropriate to tie the rulemaking effort to the accomplishment of any specific R&D plan
- The FAA is committed in their Aging Aircraft Plan to publish the Part 29 DT final rule in fiscal year 2003



## Concluding remarks ( cont. )

### On the Harmonisation aspects

- W.G.'s for Rulemaking are supported by the FAA-JAA HMT in accordance with the outcome of the 16th Annual JAA/FAA Conference and the " Better Plan for Harmonisation "
- FAA,JAA,U.S. and European Industry agree to coordinate their effort in a suitable timeframe to accommodate TOGAA recommendations without disrupting harmonisation
- JHWG will monitor progress of different W.G.'s
- Progress from R&D supportive of the W.G.'s ; coordination is a key for success.

## **SESSION G**

### **RAPPORTEURS CONCLUSIONS & SUMMARY**



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